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ANALYSIS AND DESIGN STUDY OF A PILOT ASSIST SYSTEM FOR HELICOPTERS

By

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Edward L. Warren

April 1971

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
FORT EUSTIS, VIRGINIA

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AMERICAN NUCLEONICS CORPORATION
WOODLAND HILLS, CALIFORNIA

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This report was prepared by American Nucleonics Corporation under the terms of contract DAAJ02-70-C-0019. It discusses the approach used in the analysis and design of a pilot assist system for use in light and medium size helicopters to improve their stability and control characteristics.

The object of this contractual effort was to provide a prototype pilot assist system design for the UH-1B helicopter along with a report of the analysis and detailed drawings of the system design.

In general, the design solution presented in the report is a suitable approach. The prototype configuration is lightweight and is flexible as a result of the use of extra components and the provision of modes of operation that may not be required in a production system.

The conclusions and recommendations contained herein are concurred in by this Directorate. This concurrence does not imply the practicability or endorsement of the use of such a system specifically for UH-1B aircraft. However, it is believed that the design of the pilot assist system is technically feasible for use in continued reseach efforts and/or prototype development.

The technical monitors for this contract were Mr. R. Scharpf, Mr. H. Murray, and Mr. D. Simon of the Applied Aeronautics Division of this Directorate.

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ANALYSIS AND DESIGN STUDY OF A PILOT ASSIST SYSTEM FOR HELICOPTERS

ANC 72R-14

By

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Prepared by

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for

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SUMMARY

The purpose of the work performed under this contract was to conduct an analytical investigation of advanced flight control system (AFCS) requirements for light and medium size helicopters and to design a pilot assist system based on the analytical results. The pilot assist system (PAS) design goal was to develop an AFCS that is relatively light and inexpensive and that can be readily installed in a UH-1B.

An analytical investigation was performed using a combination of digital computer simulation and design programs, paperwork analysis, breadboard/analog testing and literature research. The pilot assist system design was accomplished using a combination of breadboard/analog testing, paperwork design, computer design and paperwork analysis.

Some of the significant results of the analytical investigation are as follows:

1. First-cut pilot assist system requirements have been generated.

2. A versatile pilot assist system has been designed for further development and evaluation testing.

3. A math model of the pilot assist system/UH-1B helicopter has been developed.

4. Digital computer simulation and design programs have been developed which can be used to significant advantage in future simulation and flight test work.

Some of the significant results of the pilot assist system design are as follows:

1. A relatively lightweight and inexpensive pilot assist system has been designed.

2. The pilot assist system is flexible (i.e., with respect to control law modification) and should simplify further development and evaluation testing.

3. The system design provides for ease of testing and maintenance.

FOREWORD

This report represents the results of the efforts expended by American Nucleonics Corporation (ANC) in performance of USAAVLABS Contract DAAJ02-70-C-0019 (Task 1F162204A13905). The work was conducted from January 1970 through December 1970. Mr. Edward Warren was the ANC Program Manager and Mr. Arthur Welch was the ANC Project Engineer.

Mr. R. Scharpf was the initial USAAVLABS technical monitor during this program. His advice and technical support constituted a significant contribution to this program. Mr. H. Murray and Mr. D. Simon, subsequent USAAVLABS technical and assistant technical monitors, respectively, maintained the high level of program support that was initiated by Mr. Scharpf.

The authors gratefully acknowledge the assistance of Messrs. R. Wyllie and V. DeSantis in the design and development of the hardware.

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LIST OF SYMBOLS

Al	roll control (lateral stick deflection)(in.)
Bl	pitch control (longitudinal stick deflection)(in.)
CO	height control (collective stick deflection)(in.)
DLR	yaw control (pedal deflection)(in.)
g	acceleration due to gravity $\left(\frac{ft}{sec^2}\right)$
h	altitude perturbation (ft)
I_x, I_y, I_z	moments of inertia about x, y, and z axes (slug-ft^2)
I_{xz}	product of inertia (slug-ft^2)
j	$\sqrt{-1}$
L	rolling moment (ft-lb)
L_p	$\partial L/\partial p$
L_r	$\partial L/\partial r$
L_v	$\partial L/\partial v$
L_{Al}	$\partial L/\partial Al$
m	mass of the aircraft $\left(\frac{lb\text{-}sec^2}{ft}\right)$
M	pitching moment (ft-lb)
M_q	$\partial M/\partial q$
M_u	$\partial M/\partial u$
M_w	$\partial M/\partial w$
M_{Bl}	$\partial M/\partial Bl$
N	yawing moment (ft-lb)
N_p	$\partial N/\partial p$
N_r	$\partial N/\partial r$
N_v	$\partial N/\partial v$
N_{LLR}	$\partial N/\partial DLR$

LIST OF SYMBOLS - Continued

p	roll rate (rad/sec)
q	pitch rate (rad/sec)
r	yaw rate (rad/sec)
s	Laplace operator, $s = \sigma + j\omega$
t	time (sec)
u	perturbation velocity along x-axis (ft/sec)
u_g	gust velocity along x-axis (ft/sec)
U_o	steady state velocity along x-axis (ft/sec)
v	perturbation velocity along y-axis (ft/sec)
w	perturbation velocity along z-axis (ft/sec)
W	gross weight (lb)
x	horizontal displacement in direction of x-axis
X	force in x-direction (lb)
X_q	$\partial X/\partial q$
X_u	$\partial X/\partial u$
X_{B1}	$\partial X/\partial B1$
y	side displacement in direction of y-axis
Y	force in y-direction (lb)
Y_p	$\partial Y/\partial p$
Y_r	$\partial Y/\partial r$
Y_v	$\partial Y/\partial v$
Y_{A1}	$\partial Y/\partial A1$
Z	force in z-direction (lb)
Z_q	$\partial Z/\partial q$
Z_u	$\partial Z/\partial u$

LIST OF SYMBOLS - Continued

Z_w	$\partial Z/\partial w$
Z_{CO}	$\partial Z/\partial CO$
α	angle of attack (rad)
β	sideslip angle of attack, $\doteq \frac{v}{U_o}$ (rad)
Δ	transfer function denominator
ζ	damping ratio
θ	pitch angle (rad)
μ	tip speed ratio
σ	real part of s
τ	time delay (sec)
φ	roll angle (rad)
ϕ	yaw angle (rad)
ω	imaginary part of s
Ω	rotational speed (rad/sec)
$<$	less than
$>$	greater than
$<<$	much less than
$>>$	much greater than
∂	partial derivative
$=$	equal
$\approx$	approximately equal

INTRODUCTION

In January 1970, work was initiated by ANC to conduct an evaluation of control feedback and servo configurations for helicopters. The planned duration of the entire program (including flight test development and evaluation) was estimated to be 2 years. The vehicle that was envisioned for the flight test work was a UH-1B with a standard rotor system and without a stabilizer bar. A potential practical result of the program is a production PAS that is readily installable in a UH-1B (thereby enhancing its handling qualities) and is applicable with minor modifications to other light- and medium-size helicopters.

The purpose of the work done to date by ANC was to provide an analytical base for efficiently conducting the remainder of the program and to design flight hardware that could easily be built, flight tested and refined.

Shortly after the beginning of the program USAAVLABS supplied ANC with digital computer data that described the UH-1B (i.e., standard rotor system and no stabilizer bar) aircraft dynamics and with other aircraft descriptive data. The basic aircraft data was used by ANC to generate an analytical base and to design the flight test PAS hardware.

This program consisted of two interrelated efforts:

1. An analytical investigation.

2. A UH-1B PAS hardware design effort.

Results of the analytical investigation beginning with the system design objectives and proceeding through to analytical results are covered in the following sections. The analytical discussion is followed by a description of the hardware design.

The primary objective of this program was to set the stage for ground based simulator and/or flight testing that would further develop and refine the hardware design reported here. By virtue of this objective the analytical investigation was conducted with more emphasis placed on work that resulted in tools that are useful for further testing (e.g., serve as a data reference base against which simulator and/or flight test results could be compared). Also, the hardware design was made flexible to simplify subsequent testing.

The work that now remains in fulfilling the intent of the total program is:

1. Simulation to validate UH-1B model data and provide support to a flight test program.

2. Flight test to put the PAS as designed into a realistic operational environment for final development and evaluation of the design. The purpose of this effort is ultimately to select those PAS features which are most valuable under operational conditions. This would define the desired production system.

ANALYTICAL INVESTIGATION

SYSTEM DESIGN OBJECTIVES

The following analytical design objectives, which satisfy the program objectives, were used during this study:

1. Develop a set of system handling qualities requirements (see System Design Requirements) that correlates system requirements with flight tasks for light- and medium-size helicopters and provides the basis for a system test specification for the following types of system test setups:

   a. In-house with PAS connected to breadboard simulator.

   b. With hardware PAS or simulation connected to a ground-based simulator.

   c. With hardware PAS connected to a simulator at the aircraft for preflight checks.

   d. With hardware PAS installed in a vehicle for in-flight testing.

2. Design a "nominal" PAS system (see System Math Model) that enhances the UH-1B stability and control characteristics. The PAS should satisfy the requirements generated in item 1 and result in reducing pilot workload by providing:

   a. Vehicle stabilization in all selected modes of operation.

   b. Gust alleviation over the basic aircraft.

   c. Precision hover control.

   d. Decoupling, i.e., reducing lateral and longitudinal interaction.

   e. Cruise mode control.

   f. Automatic turn coordination.

3. A system description that can be used to:

   a. Allow the PAS to be evaluated and refined in a ground based simulation of the UH-1B.

   b. Build a PAS that can be flight evaluated and refined in a UH-1B.

4. Provide sufficient flexibility in the system design so that control laws may be altered (within reasonable limits) during either the ground-based simulation or flight test evaluation; i.e., continue to use the system as an advanced flight control system tool.

5. Provide analysis tools; i.e., digital programs and a breadboard analog system, that can be used as checks during:

   a. The in-house hardware design phase.

   b. Ground-based simulation work.

   c. Flight test evaluation and refinement.

6. Analyze and document alternate PAS system designs, i.e., combinations of control laws and servo configurations, that may be further evaluated (either in a ground-based simulation or a flight test evaluation).

7. Recommend areas for further work that will result in determining the desirability of a production PAS.

SYSTEM DESIGN REQUIREMENTS

A summary of system requirements (handling and flying qualities) versus flight tasks are tabulated in Tables I and II. By making a "Flight Task" selector panel available to the pilot, it is possible to have the system performance as indicated in Tables I and II. It is possible to evaluate this approach to system operation (either in flight or during simulator work) by setting the "Mode Selector" switches in the appropriate position for each flight task.

Tables I and II document ballpark system requirements that will be used as initial system specifications. Subsequent ground based simulation and/or flight testing will result in further development and refinement of Tables I and II.

Table III lists the system requirements in a more conventional sense, i.e., Pilot Assist System Function or Characteristic. This tabulation covers only a portion of what might be construed by an aircraft user to be full coverage of flying and handling qualities requirements which fall within the scope of this program. The following quotation (Reference 1) explains the previous sentence:

> "To the user of an aircraft, flying and handling qualities are not only the airframe dynamics, control feel, control sensitivity, etc., but all things which affect the performance of the aircraft design mission. These include such parameters as

TABLE I. SYSTEM REQUIREMENTS VS FLIGHT TASK (HOVER, AIR TAXI AND DESCENDING FLIGHT)

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - IAS Hold</u>		
1. Pitch steering and release	1. Longitudinal A/S response shall be smooth and fairly rapid. Upon release, the A/C shall be within 10% of the new trim A/S within 20 sec (maximum of one undershoot not to exceed 20%).	Control forward and lateral airspeed via cyclic stick force inputs.
2. Roll steering and release	2. Same as 1 except for lateral A/S.	
<u>IAS Hold</u>		
1. Accuracy (steady state)	1. Shall maintain the reference (ship A/S system) IAS within ±2 kt.	The standard UH-1B airspeed system is apparently accurate within ±4 kt in level flight and within ±6 kt for climbing flight.
2. Speed hold range	2. 0 to 110 kt (130 kt for 540 rotor)	
3. Transient response to 5 kt step input	3. Maximum of one overshoot not to exceed 20%. Rise time shall be no greater than 25 sec.	
4. Residual oscillations	4. Shall not exceed ±2 kt, vertical or longitudinal accelerations of ±.05 (at pilot's station), pitch attitudes ≥ ±.25 degrees, nor a period less than 20 sec.	

TABLE I - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Vertical Speed Hold</u>		
1. Collective force at vertical rates ≤ 1.5 ft/sec	1. Vertical rate is proportional to collective force. The vertical rate response to a step force change shall be smooth and have a rise time of less than 2 sec (maximum of one overshoot not to exceed 20%).	Command vertical speed (including zero) through collective force to maneuver or maintain a vertical speed (have altitude hold for low force and low vertical rate) of A/C.
2. Collective force at vertical rates > 1.5 ft/sec	2. A force ≤ 1.5 lb shall command a proportional incremental vertical rate. A force > 1.5 lb shall command a vertical acceleration.	
<u>Vertical Speed Hold</u>		
1. Accuracy (steady state)	1. Shall maintain the reference (sensor) vertical speed within ±20 ft/min.	
2. Speed hold range	2. ±2000 ft/min.	
3. Transient response to 5 ft/sec step input	3. Maximum of one overshoot not to exceed 20%. Rise time shall be no greater than 2.0 sec.	
4. Residual oscillations (steady state)	4. Not to exceed .05 g at pilot station; less than .25 degree pitch angle; period shall be no less than 20 sec.	

TABLE I - Continued

Pilot Assist System Function of Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Heading Hold</u>		
1. Pedal steering and release	1. Yaw rate response shall be smooth and rapid. A/C shall be capable of developing at least ±45 degrees/sec yaw rate through pedal command. Upon release, the A/C shall settle at a new trim heading within 20 sec (maximum of one undershoot not to exceed 20%).	Command flat turns through pedal force to change heading
<u>Heading Hold</u>		
1. Accuracy (steady state)	1. Shall maintain reference heading error within ±0.5 degree.	
2. Transient response to 0.15 g lateral acceleration input	2. Maximum of one overshoot not to exceed 20%. Rise time (10 to 90%) shall be no greater than 15 sec.	
3. Residual oscillations (steady state)	3. Not to exceed 0.1 degree of roll or yaw; period shall be no less than 10 sec.	

TABLE II. SYSTEM REQUIREMENTS VS FLIGHT TASK (ACCELERATING FLIGHT AND FORWARD FLIGHT TURNS)

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Attitude Hold</u>		
1. Pitch steering and release	1. Pitch rate response shall be smooth and rapid. A/C shall be capable of developing at least ±15 degrees/sec pitch rate through stick command. Upon release, the A/C shall settle at a new trim attitude within 3 sec (maximum of one undershoot not to exceed 20%).	Command new altitude via cyclic pitch force; lateral cyclic force commands velocity below 30 kt and roll attitude above 50 kt with blend in between.
2. Roll steering and release	2. Same as 1 except A/C shall be capable of developing ±50 degrees/sec roll rate.	
<u>Pitch and Roll Attitude Hold</u>		
1. Accuracy (steady state)	1. Shall maintain reference pitch and roll attitude to within ±0.5 degree.	
2. Transient response to 30 degrees roll or 5 degrees pitch attitude step input	2. Maximum of one overshoot not to exceed 20% of difference between initial and steady state values. Rise time (10 to 90%) shall be no greater than 2 sec.	
3. Residual oscillation (steady state)	3. At pilot's station, not to exceed ±.05 g normal acceleration, ±.02 g lateral acceleration, ±.25 degree pitch and yaw, ±.5 degree roll; period shall be no less than 10 sec	

TABLE II - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Baro Altitude Hold</u>		
1. Collective steering and release	1. Vertical rate response shall be smooth and rapid. A/C shall be capable of developing ±35 ft/sec vertical rate through stick command. Upon release, the A/C shall settle at a new trim altitude within 20 sec (maximum of one undershoot not to exceed 20%).	Command new altitude via collective force input. Hold altitude via absence of force.
<u>Barometric Altitude Hold</u>		
1. Accuracy (straight and level steady state)	1. Shall maintain reference altitude within ±5 ft or ±.1%, whichever is larger.	
2. Altitude during turns	2. Not to exceed ±20 ft or ±.3%, whichever is larger.	
3. Engagement during climb or descent	3. The altitude at the time of engagement shall be captured with maximum of one overshoot. Normal acceleration shall not exceed ±.25 G.	
4. Transient response to 50 ft step input	4. First and second overshoots shall be less than 20% and 5%, respectively. The rise time (10 to 90%) shall be no greater than 15 sec.	
5. Residual oscillation (steady state)	5. Not to exceed .05 at pilot station; less than .25 degree pitch angle; period shall be no less than 20 sec.	

TABLE II - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Heading Select</u>		
1. Pedal steering and release	1. Yaw rate response shall be smooth and rapid. A/C shall be capable of developing at least ±45 degrees/sec yaw rate through pedal command. Upon release, the A/C shall settle at a new trim heading within 20 sec (maximum of one undershoot not to exceed 20%).	Command automatic coordinated turns above 20 kt via heading bug change.
<u>Heading Select</u>		
1. Accuracy (steady state)	1. Shall maintain reference within 0.5 degree.	
2. Transient response to 0.15 g lateral acceleration input	2. Maximum of one overshoot not to exceed 20%. Rise time (10 to 90%) shall be no greater than 15 sec.	
3. Residual oscillations (steady state)	3. Not to exceed 0.1 degree of roll or yaw; period shall be no less than 10 sec.	

TABLE II - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Heading Hold</u>		
1. Pedal steering and release	1. Yaw rate response shall be smooth and rapid. A/C shall be capable of developing at least ±95 degrees/sec yaw rate through pedal command. Upon release, the A/C shall settle at a new trim heading within 20 sec (maximum of one undershoot not to exceed 20%).	Command semi-automatic coordinated turns above 20 kt via lateral cyclic force input.
<u>Heading Hold</u>		
1. Accuracy (steady state)	1. Shall maintain reference within 0.5 degree.	
2. Transient response to 0.15 g lateral acceleration input.	2. Maximum of one overshoot not to exceed 20%. Rise time (10 to 90%) shall be no greater than 15 sec.	
3. Residual oscillations (steady state)	3. Not to exceed 0.1 degree of roll or yaw; period shall be no less than 10 sec.	

TABLE III. SYSTEM REQUIREMENTS VS. PAS FUNCTION

Pilot Assist System Function or Characteristic	System Requirements	Comments
Control Force Maneuvering - IAS Hold		
1. Pitch steering and release	1. Longitudinal A/S response shall be smooth and fairly rapid. Upon release, the A/C shall be within 10% of the new trim A/S within 20 sec of release (maximum of one undershoot not to exceed 20%).	Control forward and lateral airspeed via cyclic stick force inputs.
2. Roll steering and release	2. Same as 1 except for lateral A/S.	
IAS Hold		
1. Accuracy (steady state)	1. Shall maintain the reference (ship A/S system) IAS within ±5 kt.	The standard UH-1B airspeed system is apparently within ±4 kt in level flight and within ±6 kt for climbing flight.
2. Speed hold range	2. 0 to 110 kt (130 kt for 540 rotor).	
3. Transient response to 5-kt step input	3. Maximum of one overshoot not to exceed 20%. Rise time shall be no greater than 25 sec.	
4. Residual oscillations (steady state)	4. Shall not exceed ±2 kts; nor vertical or longitudinal accelerations of ±.05 g (at pilot's station); pitch attitudes ≥ .25 degree; and a period no less than 20 sec.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Attitude Hold</u>		
1. Pitch steering and release	1. Pitch rate response shall be smooth and rapid. A/C shall be capable of developing at least ±15 degrees/sec pitch rate through stick command. Upon release, the A/C shall settle at a new trim attitude within 3 sec (maximum of one undershoot not to exceed 20%).	Command new altitude via cyclic pitch force; lateral cyclic force commands velocity below 30 kt and roll attitude above 50 kt with a blend in between.
2. Roll steering and release	2. Same as 1 except A/C shall be capable of developing ±50 degrees/sec roll rate.	
<u>Pitch and Roll Attitude Hold</u>		
1. Accuracy (steady state)	1. Shall maintain reference pitch and roll attitude to within ±0.5 degree.	
2. Transient response to 30 degrees roll or 5 degrees pitch attitude step input	2. Maximum of one overshoot not to exceed 20% of difference between initial and steady state values. Rise time (10 to 90%) shall be no greater than 2 sec.	
3. Residual oscillation (steady state)	3. At pilot's station, not to exceed ±.05 g normal acceleration, ±.02 g lateral acceleration, ±.25 degrees pitch and yaw and ±.5 degree roll; period shall be no less than 10 sec.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Vertical Speed Hold</u>		
1. Collective force at vertical rates ≤ 1.5 ft/sec	1. Vertical rate is proportional to collective force. The vertical rate response to a step force change shall be smooth and have a rise time of less than 2 sec (maximum of one overshoot not to exceed 20%).	Command vertical speed (including zero) through collective force to maneuver or maintain a vertical speed (have altitude hold for low force and low vertical rate) of A/C.
2. Collective force at vertical rates > 1.5 ft/sec	2. A force ≤ 1.5 lb shall command a proportional incremental vertical rate. A force > 1.5 lb shall command a vertical acceleration.	
<u>Vertical Speed Hold</u>		
1. Accuracy (steady state)	1. Shall maintain the reference (sensor) vertical speed within ±20 ft/min.	
2. Speed hold range	2. ±1000 ft/min.	
3. Transient response to 5 ft/sec step input	3. Maximum of one overshoot not to exceed 20%. Rise time shall be no greater than 2 sec.	
4. Residual oscillations (steady state)	4. Not to exceed .05 g at pilot station; less than .25 degree pitch angle; period greater than 20 sec.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
Control Force Maneuvering - Baro Altitude Hold		
1. Collective steering and release	1. Vertical rate response shall be smooth and rapid. A/C shall be capable of developing ±35 ft/sec vertical rate through stick command. Upon release, the A/C shall settle at a new trim attitude within 20 sec. (Maximum of one undershoot not to exceed 20%.)	Command new altitude via collective force input. Hold altitude via absence of force.
Barometric Altitude Hold		
1. Accuracy (straight and level steady state)	1. Shall maintain reference altitude within ±5 ft or ±.1%, whichever is larger.	
2. Altitude during turns	2. Not to exceed ±20 ft or ±.3%, whichever is larger.	
3. Engagement during climb or descent	3. The altitude at the time of engagement shall be the reference altitude. The reference altitude shall be captured with no overshoot. Normal acceleration shall not exceed .25 g.	
4. Transient response to 50 ft step input	4. First and second overshoots shall be less than 20% and 5%, respectively. The rise time (10 to 90%) shall be no greater than 15 sec.	
5. Residual oscillation (steady state)	5. Not to exceed .05 g at pilot station; less than .25 degree pitch angle; period ≥ 20 sec.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Heading Select</u>		
1. Pedal steering and release	1. Yaw rate response shall be smooth and rapid. A/C shall be capable of developing ±45 degrees/sec yaw rate through pedal command. Upon release, the A/C shall settle at a new trim heading within 20 sec. (Maximum of one undershoot not to exceed 20%.)	Command automatic coordinated turns above 20 kt via heading bug change.
<u>Heading Select</u>		
1. Accuracy (steady state)	1. Shall maintain reference within .05 degree.	
2. Transient response to 0.15 g lateral acceleration input	2. Maximum of one overshoot not to exceed 20%. Rise time (10 to 90%) shall be no greater than 15 sec.	
3. Residual oscillations (steady state)	3. Not to exceed 0.1 degrees of roll or yaw; period shall be no less than 10 sec.	
<u>Automatic Turn Coordination</u>		
1. Lateral acceleration during steady state bank	1. Maximum lateral acceleration of .03 g at C.G.	
2. Lateral acceleration during rolling maneuvers	2. Maximum of 0.1 g while rolling from 30 degrees on one side to 30 degrees on the other side, up to maximum roll rate.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Control Force Maneuvering - Heading Hold</u>		
1. Pedal steering and release	1. Yaw rate response shall be smooth and rapid. A/C shall be capable of developing at least ±45 degrees/sec yaw rate through pedal command. Upon release the A/C shall settle at a new trim heading within 20 sec (maximum of one undershoot not to exceed 20%).	Command semi-automatic coordinated turns above 20 kt via lateral cyclic force input.
<u>Heading Hold</u>		
1. Accuracy (steady state)	1. Shall maintain reference within 0.5 degree.	
2. Transient response to 0.15 g lateral acceleration input	2. Maximum of one overshoot not to exceed 20%. Rise time (10 to 90%) shall be no greater than 15 sec.	
3. Residual oscillations (steady state)	3. Not to exceed 0.1 degree of roll or yaw; period shall be no less than 10 sec.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Decoupling</u>		
1. Pitch rate response to .5 in. lateral cyclic and .5 in. pedal step inputs, respectively. 2. Altitude rate response to .5 in. lateral cyclic and .5 in. pedal step inputs, respectively. 3. Roll rate response to .5 in. F/A cyclic and .5 in. collective step inputs, respectively. 4. Yaw rate response to .5 in. F/A cyclic and .5 in. collective step inputs, respectively.	Maximum cross axis rates should not exceed 10% of the cross axis rates that occur for the free A/C (both angular and translational). Maximum cross axis angular rate (e.g., pitch due to roll) shall not exceed 10% of the peak in-axis (e.g., roll) rate.	
<u>Synchronization</u>		
1. Airspeed 2. Vertical rate 3. Altitude (baro or radar) 4. Pitch attitude	All synchronization loops shall have a loop gain $\geq$ 40 degree /sec/V (i.e., time constant $\leq$ .025 sec). In the hold mode, the following maximum synchronizer drift rates apply: 1. Airspeed $\leq \pm 0.1$ ft/sec/min 2. Vertical rate $\leq \pm 0.02$ ft/sec/min	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Synchronization (Continued)</u>		
5. Roll attitude	3. Altitude $\leq \pm 0.1$ ft/min 4. Pitch attitude $\leq \pm 0.005$ degree/min 5. Roll attitude $\leq \pm 0.005$ degree/min	
<u>Mode Switching Transients</u>		
1. Engagement/disengagement of AFCS or any of its modes	1. Engagement or disengagement of the AFCS or any of its modes under steady state conditions shall not result in transients in excess of ± 0.05 g at C.G. in normal acceleration and ± 1 degree in roll and pitch altitude.	
<u>Command Signal Limits</u>		
1. Roll attitude	1. In outer loop mode operation the maximum bank command shall be limited to ± 30 degrees.	
2. Pitch attitude	2. In outer loop mode operation the maximum pitch command shall be limited to ± 15 degrees.	
3. Rotor RPM	3. Rotor underspeed shall be minimized via reducing the collective position command as a function of rotor speed.	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Longitudinal Cyclic Actuator</u>		
1. Parallel actuator	1. Full authority (mechanical limiting) having bandwidth (small amplitude) ≥ 3 cps. Pilot overpower force ≤ 12 lb at grip.	
2. Series actuator	2. $\leq 15\%$ authority (mechanical limiting) with small signal B.W. ≥ 5 cps.	
3. Series/parallel actuator	3. Same as 1 and 2 individually (parallel servo must oppose a series servo hardover to maintain pitch rate at ≤ 2 degrees/sec for first 2 sec).	
<u>Lateral Cyclic Actuator</u>		
1. Parallel actuator	1. Full authority (mechanical limiting) having B.W. ≥ 3 cps. Pilot overpower force ≤ 12 lb at grip.	
2. Series actuator	2. $\leq 15\%$ authority (mechanical limiting) with small signal B.W. ≥ 5 cps.	
3. Series/parallel actuator	3. Same as 1 and 2 individually (parallel servo must oppose a series servo hardover to maintain roll rate ≤ 3 degrees/sec for first 2 sec).	

TABLE III - Continued

Pilot Assist System Function or Characteristic	System Requirements	Comments
<u>Tail Rotor Actuator</u>		
1. Parallel actuator	1. Full authority (mechanical limiting) having B.W. ≥ 3 cps. Pilot over-power force $\leq$ [illegible]0 lb at pedal.	
2. Series actuator	2. $\leq 15\%$ authority (mechanical limiting) with small signal B.W. ≥ 5 cps.	
3. Series/parallel actuator	3. Same as 1 and 2 individually (parallel servo must oppose a series servo hardover to maintain yaw rates ≤ 3 degrees/sec for first 2 sec).	
<u>Collective Pitch Actuator</u>		
1. Parallel actuator	1. Full authority (mechanical limiting) having B.W. ≥ 3 cps. Pilot over-power force ≤ 15 lb at stick.	
<u>Electrical Trim System</u>		
1. Longitudinal cyclic	1. The pitch trim (via washed-out follow-up) shall be capable of following all flight configuration changes to preclude excessive mode switching transients.	
2. Collective	2. Same as 1.	
3. Lateral cyclic	3. Same as 1 , except washed out follow-up only if required.	
4. Pedal	4. Same as 1.	

pilot workload, aircraft performance, cockpit environment, instruments and displays, cockpit controllers, avionics, environmental operating conditions, etc."

Table III was generated using the following assumptions:

1. The criteria should be in a form that can be readily used to test a system for compliance (therefore, the use of time domain criteria and precise input stimuli).

2. The criteria should form a set of system (PAS/aircraft) requirements.

3. The set of system requirements should be mission or task oriented and be capable of fulfilling all significant mission or task requirements.

Table III documents ball-park system requirements that will be used as initial system specifications. Subsequent ground-based simulation and/or flight testing will result in further development and refinement of Table III.

PAS DESIGN METHODOLOGY

The basic design methodology that was used in arriving at the present design can be best described by the following sequential design steps:

1. Check the basic aircraft data* by:

   a. Generating the aircraft equations of motion from USAAVLABS-supplied digital data.

   b. Comparing American Nucleonics Corporation's open-loop transfer functions (via a root locus digital program) with those supplied by USAAVLABS.

   c. Comparing with ANC's time responses using a Continuous System Modeling Program (CSMP).

2. Generate system handling qualities requirements using available literature summarizing recent R & D results. The handling qualities requirements then serve as system design goals.

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*Since there were no UH-1B flight test data available to ANC until just recently, the normal process of checking the aircraft model against flight test data was bypassed.

3. Develop and utilize simplified (i.e., linearized and assuming higher order dynamics act as straight gains) root locus and CSMP programs interactively* to:

   a. Arrive at first cut system gains and time constants.

   b. Observe the trends resulting as gains and time constants are varied.

4. Build and utilize a hardware replica of the system using as much actual hardware as possible. In this case the pseudohardware system consisted of the following:

   a. Breadboard pitch and collective channels of PAS major loop electronics.

   b. Breadboard minor loop electronics.

   c. Hardware servo actuator and load stand (including stick grip with force sensor for simulating parallel servo operation).

   d. Analog simulation of aircraft longitudinal equations of motion.

5. Generate and utilize more complex CSMP programs that provide a near one-to-one simulation of the actual system. These CSMP programs include:

   a. PAS control nonlinearities.

   b. PAS level switching and the associated logic.

   c. PAS mode switching (both automatic and duplicating mode selector switching).

   d. Loop and mode time response check runs in the same sequence and with the same inputs that would be used in flight testing or simulator testing of a "nominal" system.

---

*This consisted of closing the loop by root locus, picking the loop gain and checking time response with CSMP. If time response was not satisfactory, the design iteration with another run with the root locus program was repeated.

PILOT ASSIST SYSTEM OPERATION

The pilot assist system (PAS) allows the pilot to control the aircraft by interacting with the automatic control functions of the PAS. The pilot enters the system by applying force inputs into the cyclic and collective sticks and pedals. The configuration of the system that he controls is determined by the level of his force input, the mode selector switch settings that exist at the time of his force input, and the magnitude of certain aircraft motions that exist at the time of his force input.

The PAS design, with externally mounted accessible potentiometers, can provide a means for quickly varying control system response. Together with suitably located electrical test input points in the PAS and an external pulse generator, the PAS design can provide an efficient research and development flight test tool. The PAS has been designed to control the aircraft either by sensed aircraft motion signals or by commands from a Flight Director Computer (FDC). In the FDC mode the PAS is essentially a control augmentation system (CAS) that accepts FDC commands.

All axes contain a pilot input force sensor, selectable servo configuration (except collective, which is only parallel) that may be either series, parallel or series/parallel and operate from hover, through transition, to cruise without perceptible transients. General performance characteristics are as follows:

1. Roll Axis - During the hover and slow taxi regime, a lateral cyclic stick force input commands a proportional velocity similar to the pitch axis. During cruise, a force input commands a proportional attitude. These functions are automatically blended at intermediate velocities.

2. Pitch Axis - Throughout the flight envelope, longitudinal cyclic stick force inputs provide for incremental changes in velocity, approximating a force proportional to velocity system. The system maintains the velocity established at the time the pilot removes the force input during the cruise regime.

3. Collective Axis - A collective stick force input commands a proportional vertical rate or vertical acceleration at all forward airspeeds. The system returns to zero vertical rate (and holds an altitude) when the pilot removes his force input when operating at low

vertical rates. The system maintains the vertical rate at the time of pilot force removal when operating at high vertical rates.

4. Yaw Axis - A pedal force input commands yaw rate. The system maintains the heading established at the time the pilot removes his force input. A coordinated turn can be obtained in the HDG HLD mode by obtaining a bank angle through lateral cyclic input. In HDG SEL mode, an automatic coordinated turn is obtained (above a certain speed) by selecting a new heading on the horizontal situation indicator (HSI).

<u>Pitch Axis PAS Operation</u>

<u>Velocity Hold Mode</u>

A block diagram of the pitch axis of the PAS is shown in Figure 1. A drawing of the mode selector switch layout is shown in Figure 2. The mode selector switches that control the PAS configuration in the pitch axis are:

1. "ENGAGE" - "OFF" (Pitch Axis Engage)

2. "ATT" - "VEL"

3. "FDC" - "PAS"

4. "SER" - "S/P" - "PAR"

For "normal" operation the above mode selector switches are set for "ENGAGE", "VEL", "PAS", and "S/P". With these settings (defined here as Velocity Hold Mode), pilot force commands a proportional incremental change in aircraft forward velocity. Upon release of this force input, the system will attain a steady-state velocity, which is the aircraft velocity that existed at the time the force was released.

The pitch axis parallel servo loop position is controlled in the air by washed-out servo position feedback. This provides an automatic trim capability; i.e., the pilot does not have to hold a force to maintain various aircraft trim airspeed conditions. A second advantage of the washed out follow-up is that there will be virtually zero steady-state airspeed error (if steady-state pitch attitude is not fed back as an inner loop closure). On the ground, the servo position feedback is converted to a straight gain, via skid switches. This is done since washed-out follow-up would cause the servo to drive

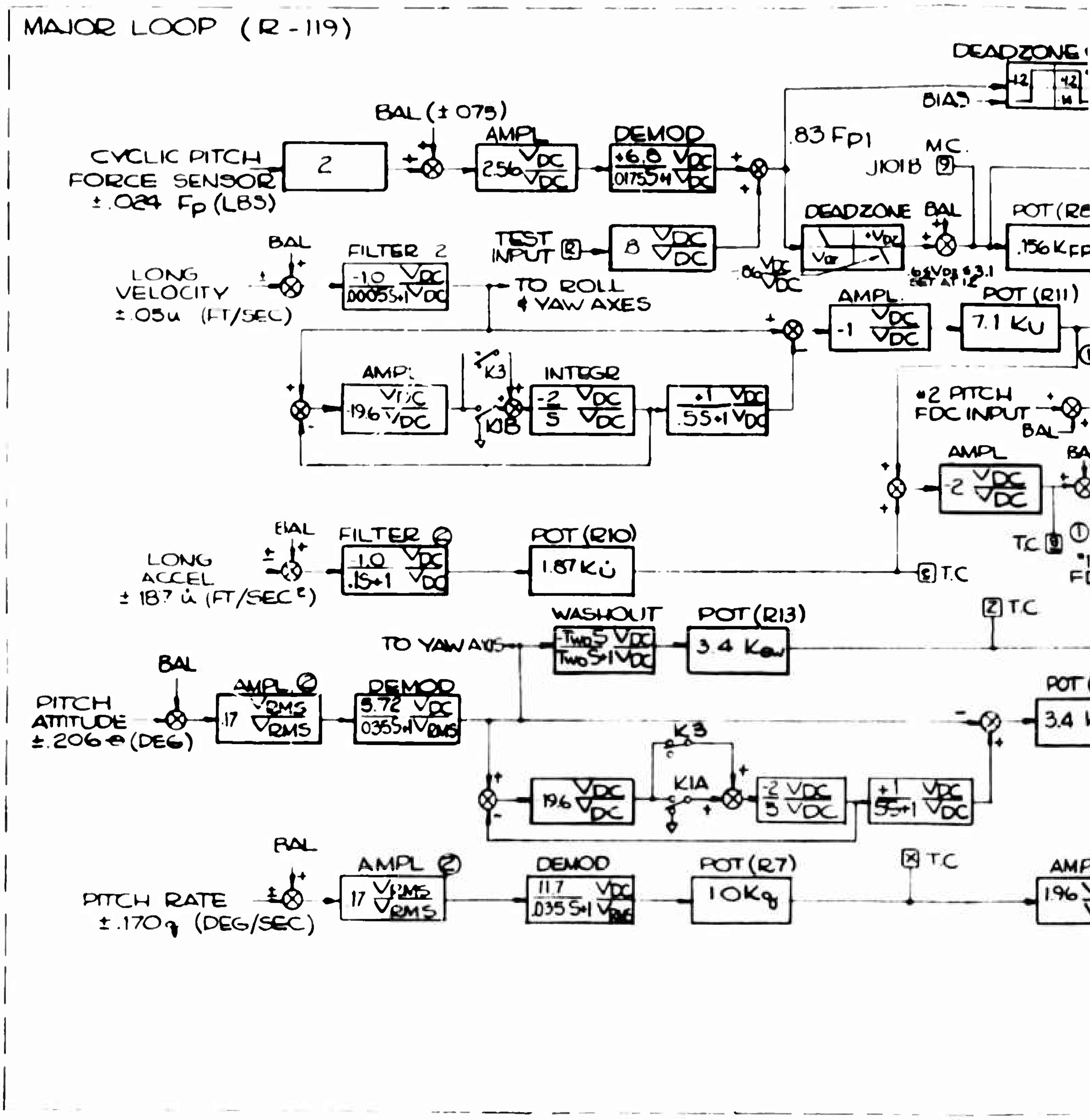
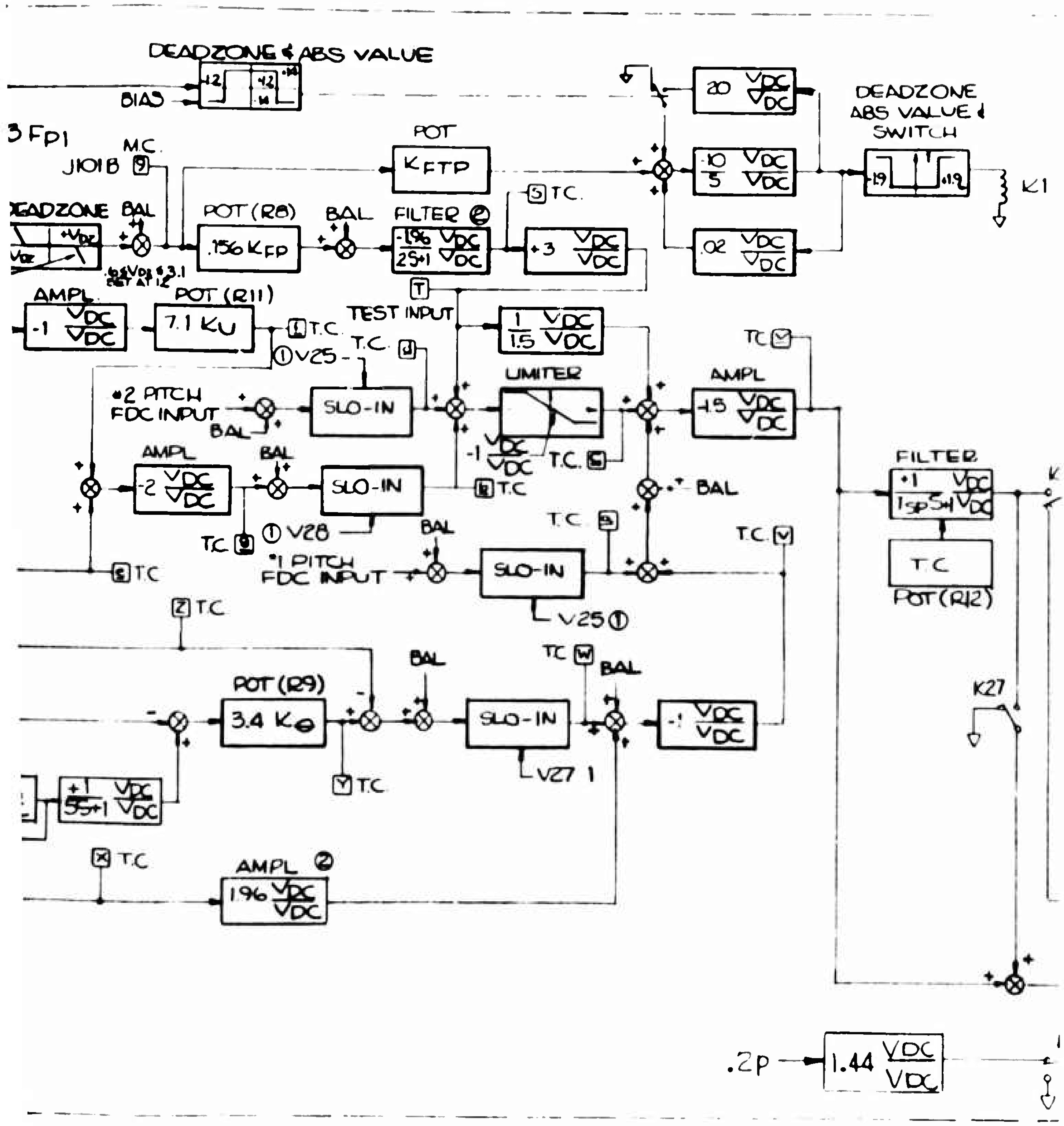
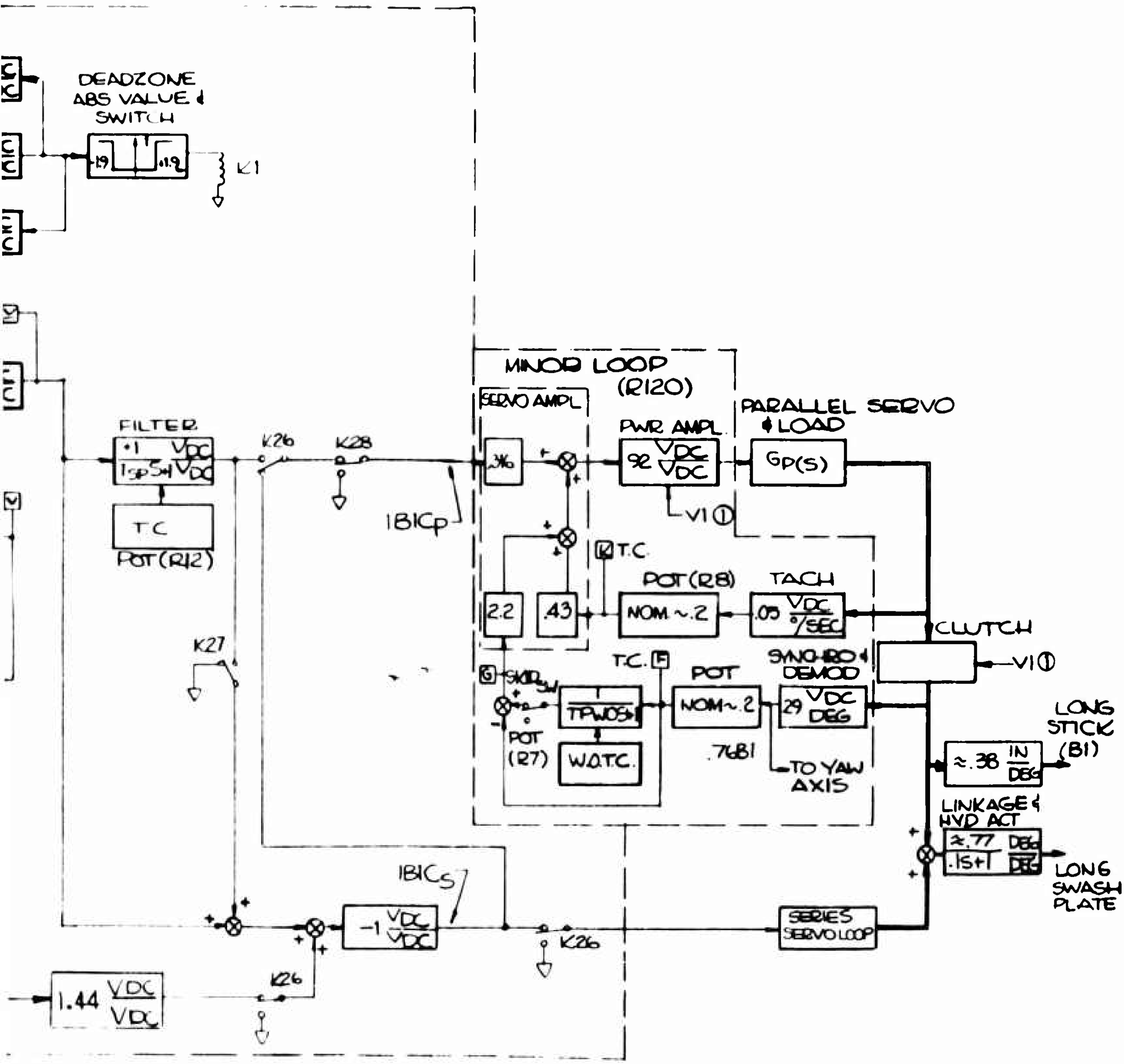


Figure 1. Pitch Axis Block Diagram



DEADZONE & ABS VALUE
BIAS
3 FP1
M.C.
J101B 9
POT
K_{FTP}
DEADZONE
BAL
POT (R8)
.156 K_{FP}
BAL
FILTER ②
5 T.C.
+3 VDC/VDC
DEADZONE ABS VALUE & SWITCH
20 VDC/VDC
.10/.5 VDC/VDC
.02 VDC/VDC
K1
AMPL
-1 VDC/VDC
POT (R11)
7.1 K_U
T.C.
① V25
TEST INPUT
T.C.
1/1.5 VDC/VDC
T.C.
LIMITER
AMPL
-1.5 VDC/VDC
#2 PITCH FDC INPUT
BAL
SLO-IN
-1 VDC/VDC
T.C.
AMPL
-2 VDC/VDC
BAL
SLO-IN
T.C.
BAL
FILTER
T.C.
POT (R12)
T.C.
① V28
BAL
T.C.
T.C.
#1 PITCH FDC INPUT
SLO-IN
V25 ①
T.C.
Z T.C.
BAL
T.C. W
BAL
K27
POT (R9)
3.4 K_θ
SLO-IN
-1 VDC/VDC
Y T.C.
V27 1
X T.C.
AMPL ②
1.96 VDC/VDC
.2P
1.44 VDC/VDC


DEADZONE
ABS VALUE &
SWITCH
.19
.19
K1
FILTER
TC
POT (R12)
K26
K28
K27
IBICp
MINOR LOOP
(R120)
SERVO AMPL
.316
2.2
.43
PWR AMPL
92 VDC/VDC
V1
PARALLEL SERVO
& LOAD
Gp(S)
T.C.
POT (R8)
NOM ~ .2
TACH
.05 VDC/°/SEC
CLUTCH
V1
SYNCHRO &
DEMOD
29 VDC/DEG
POT
NOM ~ .2
.7681
TO YAW
AXIS
T.C.
SKID SW
POT
(R7)
TPW0S+1
W.O.T.C.
LONG
STICK
(B1)
≈.38 IN/DEG
LINKAGE &
HYD ACT
≈.77 DEG/DEG
.15+1
LONG
SWASH
PLATE
IBICs
−1 VDC/VDC
K26
SERIES
SERVO LOOP
1.44 VDC/VDC
K26

C

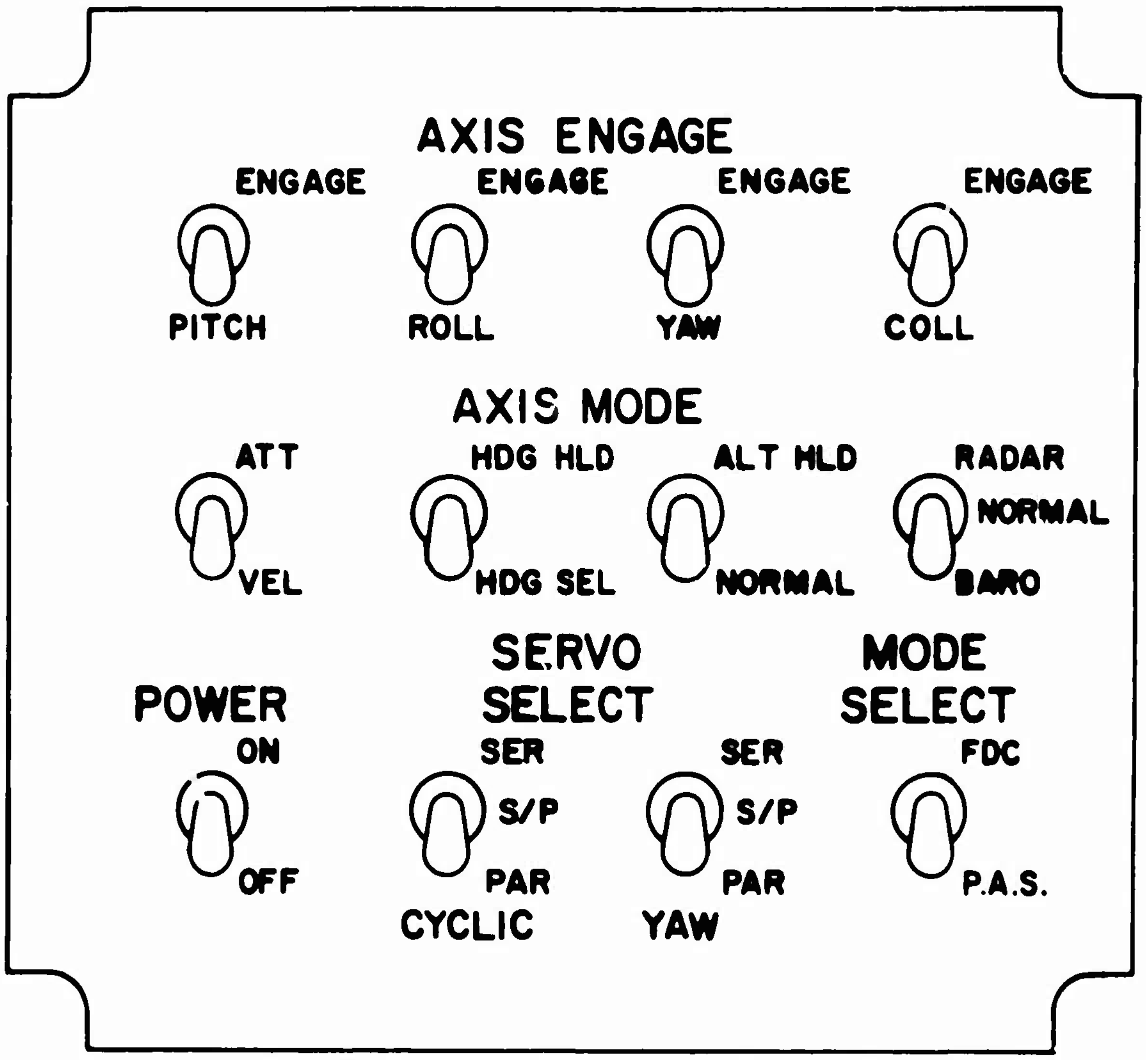


Figure 2. Mode Selector Panel Layout.

toward one of its mechanical stops (with the closed-loop servo system acting as an integrator). The command signal to the parallel servo is lagged to reduce the high-frequency feedback signals that would be felt by the pilot. The same command signal is washed out (same time constant as parallel servo lag) before becoming a series servo command signal.

The PAS signals that are available for use during the velocity hold mode (see Figure 1) are:

1. Pitch rate feedback - to suitably shape the short-period (high-frequency) dynamics of the basic aircraft.

2. Synchronized longitudinal velocity - to provide an airspeed error signal (using total velocity as an input to the synchronizer). The airspeed error is forced to approximately zero in steady state by virtue of the closed-loop integrating action of the servo loop. The airspeed error signal consists of an airspeed command signal (adjusted by the pilot by his force inputs and held when he removes his force) and a signal proportional to the aircraft's actual velocity.

3. Longitudinal acceleration - to suitably damp the aircraft's long-period response.

4. Pilot electrical force input - to provide a flexible command input into the system.

Pitch Attitude Hold Mode

The pitch attitude hold mode is an optional mode wherein pilot force commands a proportional incremental change in aircraft pitch attitude. Upon release of this force input, the system will attain a steady-state pitch attitude which is the aircraft attitude that existed at the time the force was released. For attitude hold operation, the mode selector switches are set for ENGAGE, ATT, and PAS.

The PAS signals that are used during the attitude hold mode (see Figure 1) are:

1. Pitch rate feedback - for short-period damping.

2. Synchronized pitch attitude - to provide an attitude error signal (using total pitch attitude as an input to the synchronizer). The attitude error is forced to approximately zero in steady state by virtue of the closed-loop integrating action of the servo loop. The attitude error signal consists of an attitude command signal (adjusted by the pilot via his force inputs and held when he removes his force) and a signal proportional to the aircraft's actual attitude.

3. Pilot electrical force input - to provide a flexible command input into the system.

Pitch Axis Mode Selector Switching

The PAS is designed to minimize switching transients which occur when mode selector switches are thrown. The following transient suppression actions take place when typical mode selector switches are thrown:

1. Pitch Axis Engage - From "OFF" to "ENGAGE."

   "FDC" - "PAS" switch - In "FDC", the FDC inputs are slowed in; in "PAS", either the velocity (VEL) or attitude (ATT) inputs are slowed in.

2. Pitch Axis Engage - From "ENGAGE" to "OFF" no transients occur since the servo is declutched.

3. "FDC" - "PAS" switch from "FDC" to "PAS."

   For "ENGAGE" and "ATT" settings - FDC input faded out and attitude signal faded in.

4. "FDC" - "PAS" switch from "PAS" to "FDC."

   a. For "ENGAGE" and "ATT" settings - Attitude input faded out and FDC input faded in.

   b. For "ENGAGE" and "VEL" settings - Airspeed inputs faded out and velocity signals faded in.

   c. For "OFF" and either "ATT" or "VEL" settings - No transients.

5. "ATT" - "VEL" switch from "ATT" to "VEL."

   a. For "ENGAGE" and "PAS" - Attitude signal faded out and velocity signals faded in.

   b. For "ENGAGE" and "FDC" - No effect.

   c. For "OFF" and either "FDC" or "PAS" - No effect.

6. "ATT" - "VEL" switch from "VEL" to "ATT."

   a. For "ENGAGE" and "PAS" - Velocity signals faded out and attitude signal faded in.

b. For "ENGAGE" and "FDC" - No effect.

c. For "OFF" and either "FDC" or "PAS" - No effect.

Pitch Axis Parameter Changes

Parameters associated with both the PAS major loop electronics computations and minor loop electronics are adjustable. These parameters provide the capability for rapid simulator, in-flight, or preflight adjustment. The following pitch axis parameters are readily adjustable:

1. PAS Major Loop Electronics

   a. Pitch rate gain (R7)

   b. Synchronized pitch attitude gain (R9)

   c. Washed-out pitch attitude gain (R13)

   d. Longitudinal acceleration gain (R10)

   e. Synchronized longitudinal velocity gain (R11)

   f. Pitch force gain (R8)
   g. Servo command lag and washout time constant (R12)

2. Minor Loop Electronics

   a. Tachometer feedback gain (R8)

   b. Follow-up washout time constant (R7)

Collective Axis PAS Description

General Mode Control

A block diagram of the collective axis of the PAS is shown in Figure 3. The mode selector switches (see Figure 2) which control the PAS configuration in the collective axis are:

1. "ENGAGE" - "OFF" (Collective Axis Engage)

2. "FDC" - "PAS"

3. "ALT HLD" - "NORMAL"

4. "BARO" - "NORMAL" - "RADAR"

For "normal" operation, the above mode selector switches are set for "ENGAGE", "PAS", either "ALT HLD" or "NORMAL" and

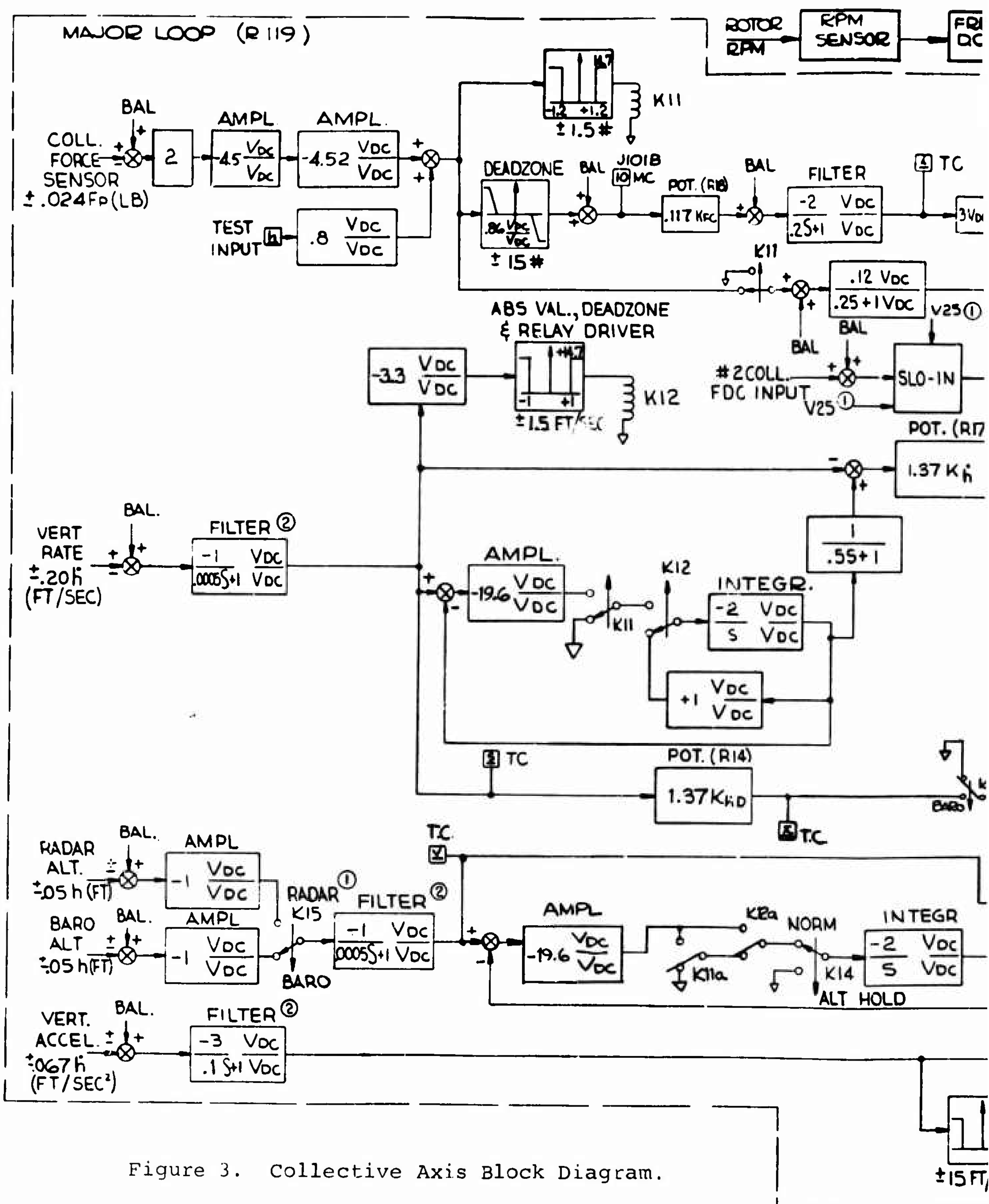


Figure 3. Collective Axis Block Diagram.

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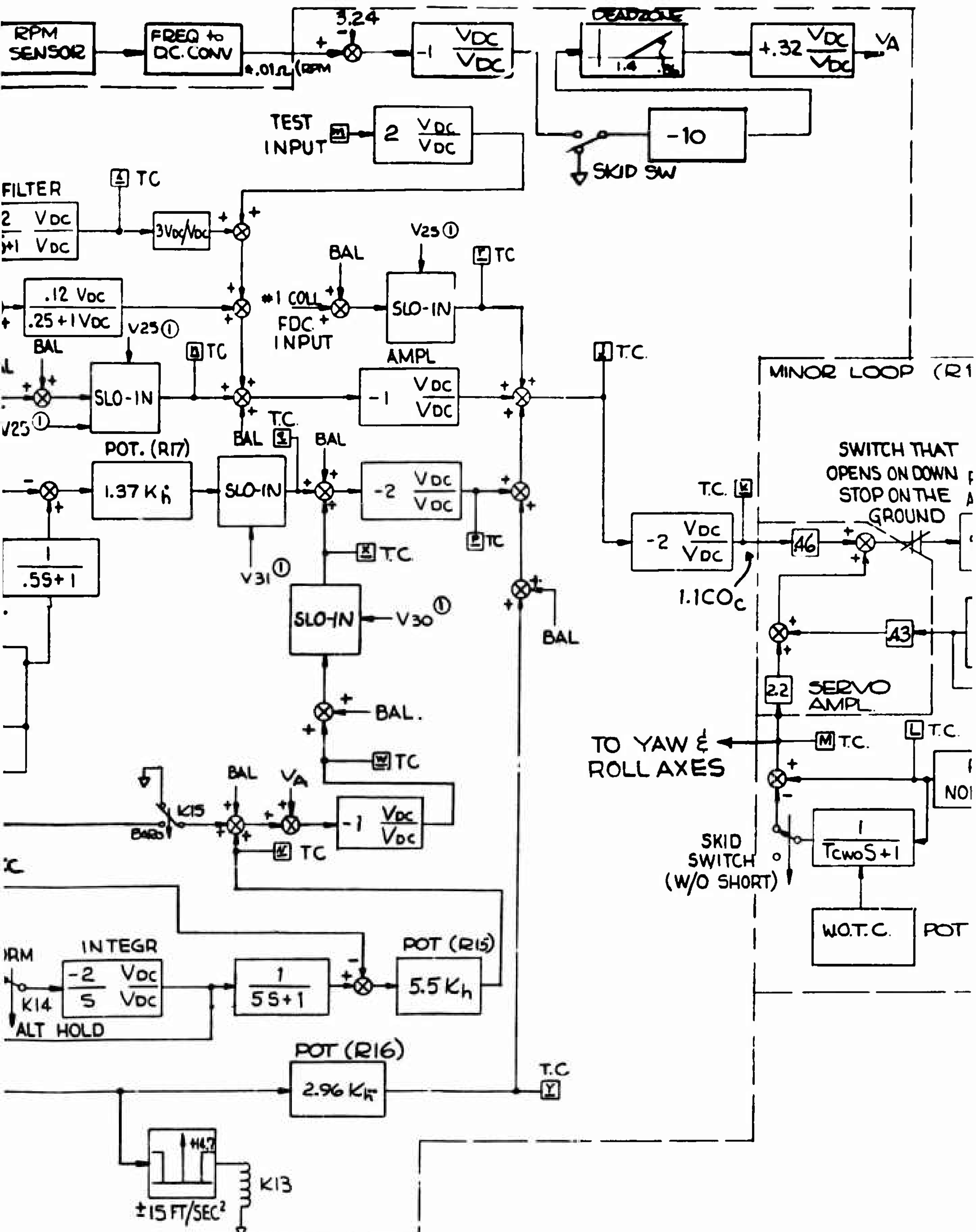
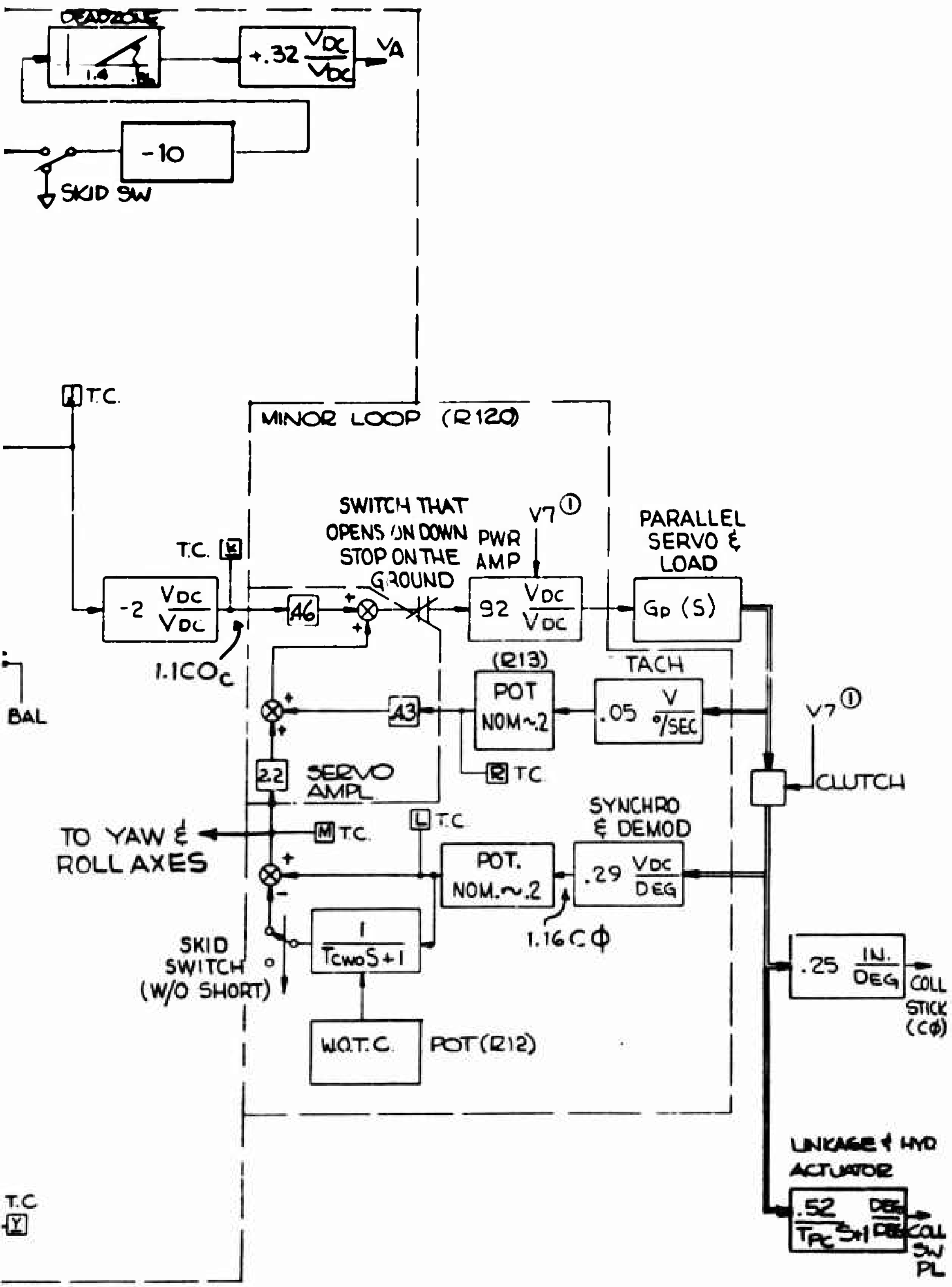


RPM SENSOR
FREQ to DC CONV
.01 (RPM
3.24
-1 VDC/VDC
DEADZONE
1.4
+.32 VDC/VDC
VA
-10
SKID SW
TEST INPUT
2 VDC/VDC
FILTER
TC
3Vdc/Vdc
V25
BAL
.12 VDC / .25+1 VDC
#1 COLL FDC INPUT
SLO-IN
AMPL
-1 VDC/VDC
T.C.
MINOR LOOP
POT. (R17)
1.37 K
-2 VDC/VDC
1 / .5S+1
V31
SLO-IN
V30
BAL.
SWITCH THAT OPENS ON DOWN STOP ON THE GROUND
1.1COc
46
43
2.2
SERVO AMPL
TO YAW & ROLL AXES
K15
BARO
VA
SKID SWITCH (W/O SHORT)
1 / TcwoS+1
W.O.T.C.
POT
INTEGR
-2 VDC / S VDC
K14
ALT HOLD
1 / 5S+1
POT (R15)
5.5 Kh
POT (R16)
2.96 Kh
K13
±15 FT/SEC²



DEADZONE
1.4
+.32 VDC/VDC
VA
-10
SKID SW
T.C.
MINOR LOOP (R120)
SWITCH THAT OPENS ON DOWN STOP ON THE GROUND
V7 ①
PWR AMP
PARALLEL SERVO & LOAD
T.C.
-2 VDC/VDC
A6
92 VDC/VDC
Gp (S)
1.1CO c
(R13)
TACH
POT NOM~.2
.05 V/°/SEC
BAL
A3
R T.C.
V7 ①
2.2
SERVO AMPL
CLUTCH
SYNCHRO & DEMOD
TO YAW & ROLL AXES
M T.C.
L T.C.
POT. NOM.~.2
.29 VDC/DEG
1.16 CΦ
SKID SWITCH (W/O SHORT)
1/TcwoS+1
.25 IN./DEG
COLL STICK (CΦ)
W.O.T.C.
POT (R12)
LINKAGE & HYD ACTUATOR
.52/TPC S+1 DEG/DEG
COLL SW PL
T.C
Y

either "BARO" or "RADAR." The choice between "BARO" and "RADAR" is simply a choice of which sensor is used as an altitude reference. The PAS configuration and operation when in either "ALT HLD" or "NORMAL" (defined here as Normal Mode) are explained in the following sections.

Normal Mode (Vertical Rate)

In the Normal Mode, the pilot controls the vertical motion and position of the aircraft by applying a force to the collective stick. The aircraft's basic vertical damping is augmented by feeding back normal acceleration to collective position. Also, a cross-feed circuit senses rotor underspeed and feeds in a correction signal to automatically lower the collective pitch. Combinations of the magnitude of his force input and the sensed vertical rate of the aircraft result in his force input producing one of the following normal mode variations:

1. Low Rate Normal Mode - When the aircraft has a vertical rate of less than 90 fpm, the collective force commands a proportional vertical rate. This mode will be referred to as the low rate mode. This mode allows the pilot to make slow vertical rate maneuvers, to make small corrections in altitude, or to hold an altitude. For this mode the aircraft is stabilized by normal acceleration and vertical rate feedbacks. By washing out the servo follow-up, the aircraft's vertical rate will be proportional to applied force in unaccelerated climbs or descents (i.e., in steady-state climbs or descents). From this mode the pilot can maneuver into the high-rate mode by applying a large collective force (thereby establishing a large vertical rate).

2. High Rate Normal Mode - The high-rate mode allows the pilot to command high vertical rate maneuvers or to accelerate (vertically) about a high vertical rate (thereby adjusting his desired high vertical rate). For this mode the aircraft is stabilized by normal acceleration for large force inputs (therefore, force commands normal acceleration) and by both normal acceleration and vertical rate for small force inputs.

   a. When the aircraft has a vertical rate of greater than 90 fpm, a large collective force (greater than 1.5 lb) commands a vertical acceleration of the aircraft.

b. When the aircraft has a vertical rate of greater than 90 fpm, a small collective force (less than 1.5 lb) holds the previous vertical rate.

Altitude Hold Mode

In the altitude hold mode, the pilot can maintain a constant altitude (by keeping his force input at less than 1.5 lb) or can maneuver about a constant altitude by applying a force greater than 1.5 lb (whereby incremental altitude is proportional to pilot force in excess of 1.5 lb). The PAS feedbacks that are used in the altitude hold mode are barometric altitude or radar altitude (which are synchronized when not in altitude hold), vertical rate, normal acceleration, and a lagged vertical rate signal that is used to filter the barometric altitude signal.

Collective Axis Parameter Changes

Parameters associated with both the PAS major loop electronics computations and minor loop electronics are adjustable. These parameters provide the capability for rapid simulator, in-flight, or preflight adjustment. The following collective axis parameters are readily adjustable:

1. PAS Major Loop Electronics

   a. Vertical acceleration gain (R16)

   b. Altitude gain (R15)

   c. Lagged vertical velocity gain (R14)

   d. Synchronized vertical velocity gain (R17)

   e. Collective force gain (R18)

2. Minor Loop Electronics

   a. Tachometer feedback gain (R13)

   b. Follow-up washout time constant (R12)

Yaw Axis PAS Description

General Mode Control

A block diagram of the yaw axis of the PAS is shown in Figure 4. The mode selector switches (see Figure 2) which control the PAS configuration in the yaw axis are:

1. "ENGAGE" - "OFF" (Yaw Axis Engage)
2. "FDC" - "PAS"
3. "HDG HLD" - "HDG SEL"
4. "SER" - "S/P" - "PAR"

For "normal" operation the above mode selector switches are set for "ENGAGE", "PAS", "S/P", and either "HDG HLD" or "HDG SEL." The PAS configuration and operation when in either the Heading Hold ("HDG HLD") mode or the Heading Select ("HDG SEL") mode are described in the following sections.

Heading Hold Mode

In the Heading Hold Mode the pilot controls the aircraft heading (at speeds below approximately 20 knots and bank angles less than 10 deg. by applying a force to the pedals. When pilot force exceeds the preselected level, the heading signal is synchronized (external to the PAS) and the pilot force commands yaw rate. The system then will maintain the heading which was established at the time the pilot removes his pedal force input.

For lateral cyclic inputs at forward speeds below approximately 20 knots and bank angles less than 10 deg. the aircraft will maintain heading unless the pilot applies a pedal force larger than the threshold.

At forward speeds above approximately 20 knots, the pilot commands a coordinated turn by banking (via lateral cyclic) the aircraft to the desired bank angle. The yaw input to coordinate the turn is computed and is used to command the proper tail rotor deflection after the bank angle has exceeded 10 deg. The pilot exits the turn by using lateral cyclic to return the aircraft to zero roll attitude. The aircraft will then hold the heading that existed when the aircraft rolled through 10 deg on the way to zero roll attitude.

Heading Select Mode

In the Heading Select Mode, the pilot controls the aircraft in heading with a turn knob (which is a part of the HSI system). The resulting heading select error commands an

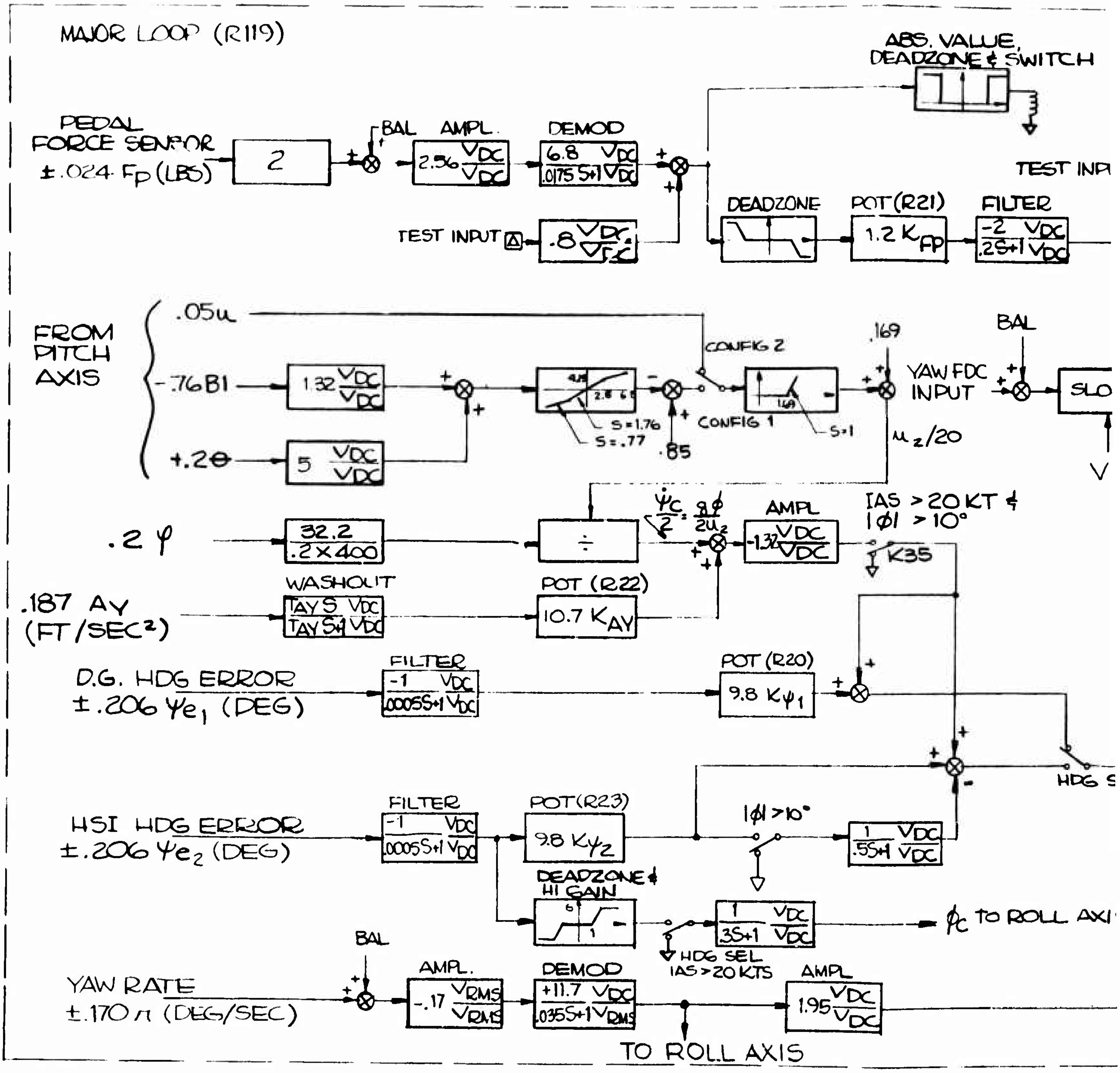


Figure 4. Yaw Axis Block Diagram

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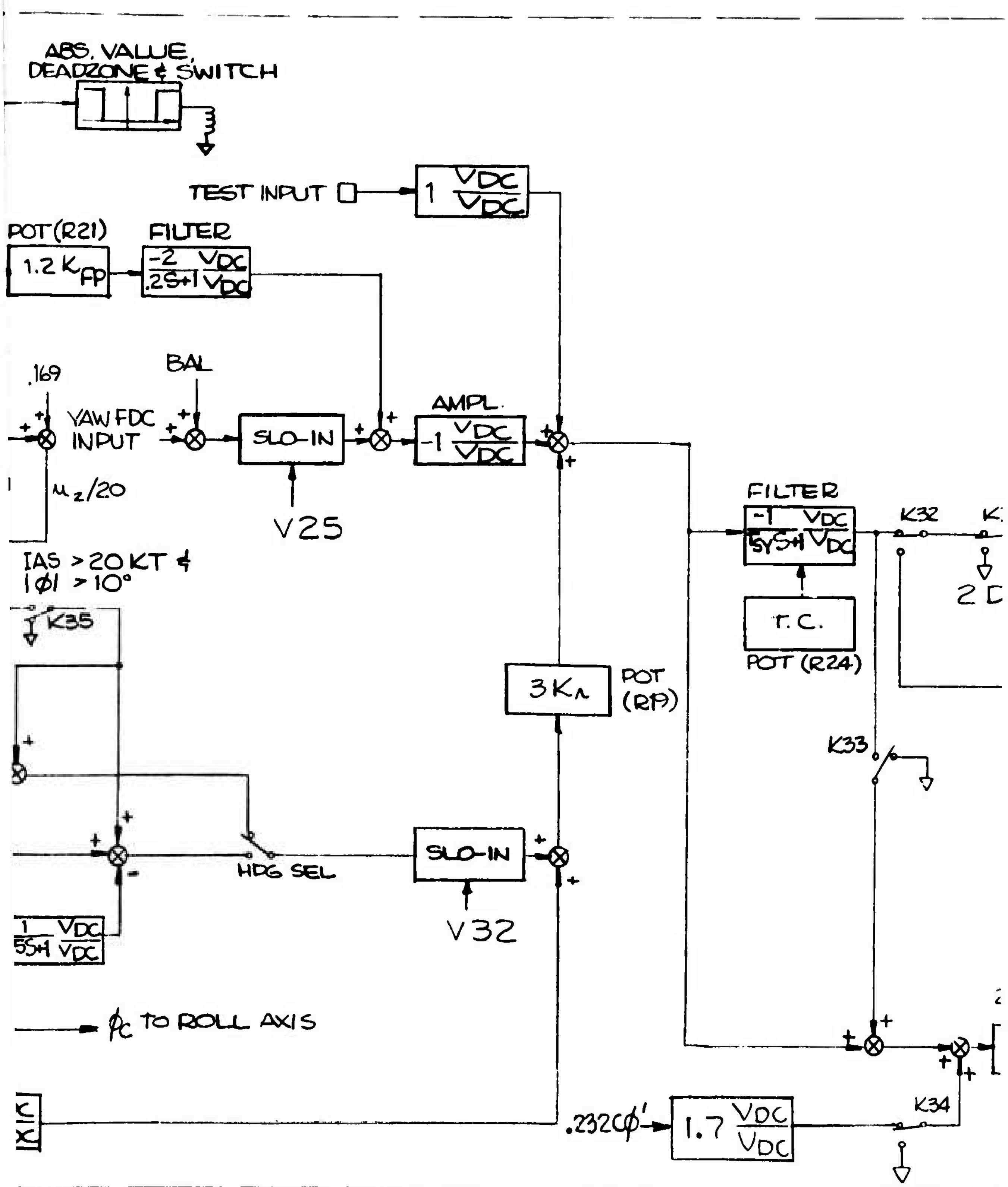
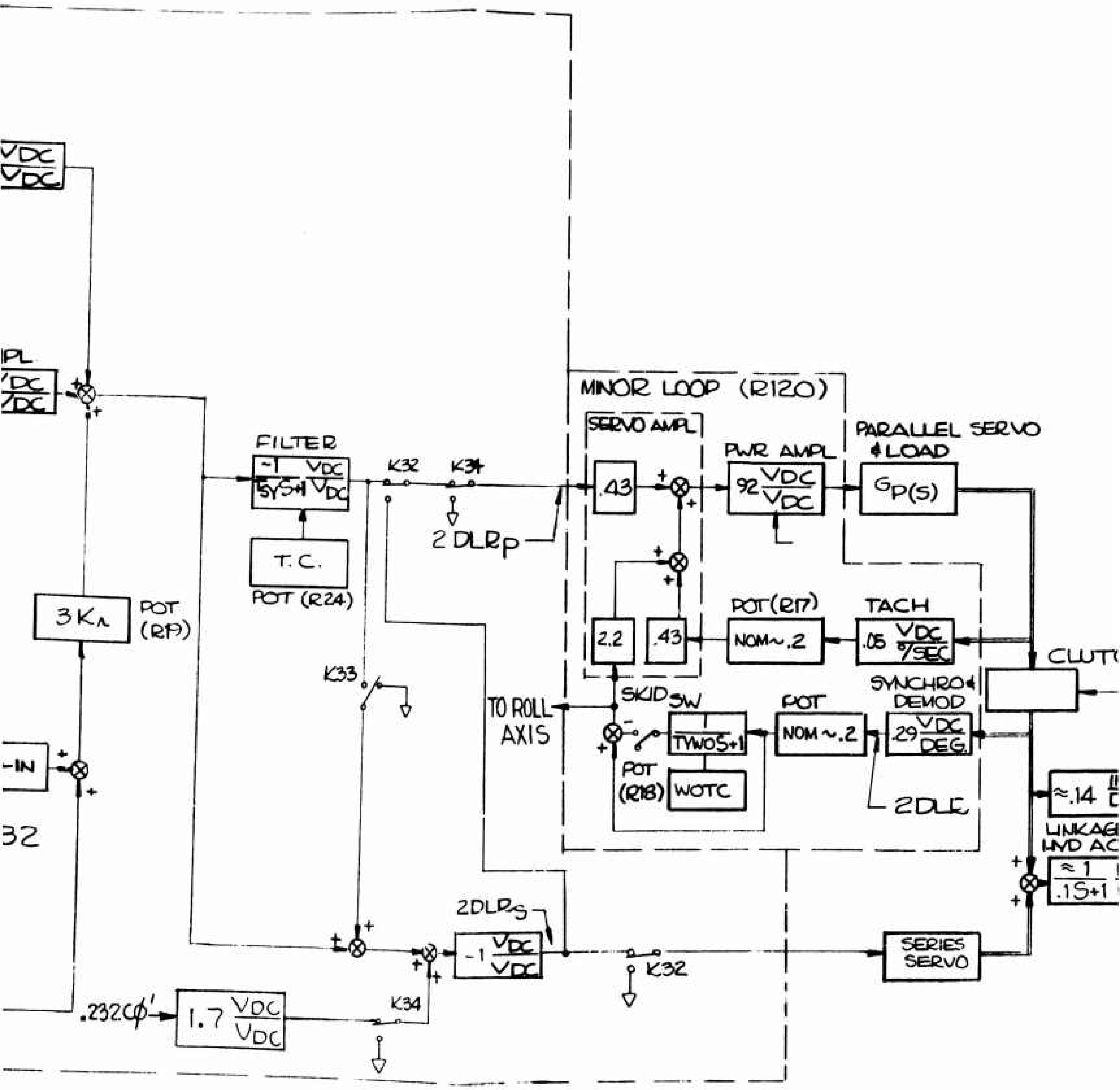

ABS. VALUE, DEADZONE & SWITCH
TEST INPUT
1 VDC/VDC
POT (R21)
1.2 K_FP
FILTER
-2/.2S+1 VDC/VDC
.169
BAL
YAW FDC INPUT
SLO-IN
AMPL.
-1 VDC/VDC
μz/20
V25
FILTER
-1/5S+1 VDC/VDC
K32
T.C.
POT (R24)
IAS > 20 KT &
|φ| > 10°
K35
3 K_λ
POT (RP)
K33
HDG SEL
SLO-IN
V32
1/5S+1 VDC/VDC
φc TO ROLL AXIS
.232Cφ'
1.7 VDC/VDC
K34

B



MINOR LOOP (R120)
SERVO AMPL
PWR AMPL
PARALLEL SERVO & LOAD
FILTER
T. C.
POT (R24)
K32
K34
2 DLRp
3 Kλ
POT (RP)
K33
TO ROLL AXIS
SKID SW
POT (R18)
WOTC
POT (R17)
NOM ~ .2
TACH
SYNCHRO & DEMOD
POT
2DLR
2DLRs
SERIES SERVO
CLUTC
LINKAGE HYD AC
.232C φ'

B

C

automatic coordinated turn above approximately 20 knots and a flat turn below that speed. Below 20 knots forward speed a constant turn rate is commanded. The pilot can add to or subtract from this constant turn rate by applying a pedal force in the appropriate direction. Upon reaching the selected heading, the system will then stabilize about that heading.

Above 20 knots forward speed a constant bank angle (30 deg.) command is generated when a new heading is selected. The proper turn rate command will then be generated in the yaw axis to obtain a coordinated turn. The pilot can fly at a bank angle other than the commanded bank angle by applying a lateral cyclic force.

Yaw Axis Parameter Changes

Parameters associated with both the PAS major loop electronics computations and minor loop electronics are adjustable. These parameters provide the capability for rapid simulator, in-flight, or preflight adjustment. The following yaw axis parameters are readily adjustable:

1. PAS Major Loop Electronics
   a. Yaw rate gain (R19)
   b. Heading hold gain (R20)
   c. Pedal force gain (R21)
   d. Lateral acceleration gain (R22)
   e. Heading select gain (R23)
2. Minor Loop Electronics
   a. Tachometer feedback gain (R17)
   b. Follow-up washout time constant (R18)

Roll Axis PAS Description

General Mode Control

A block diagram of the roll axis of the PAS is shown in Figure 5. The mode selector (see Figure 2) switches that control the PAS configuration in the roll axis are:

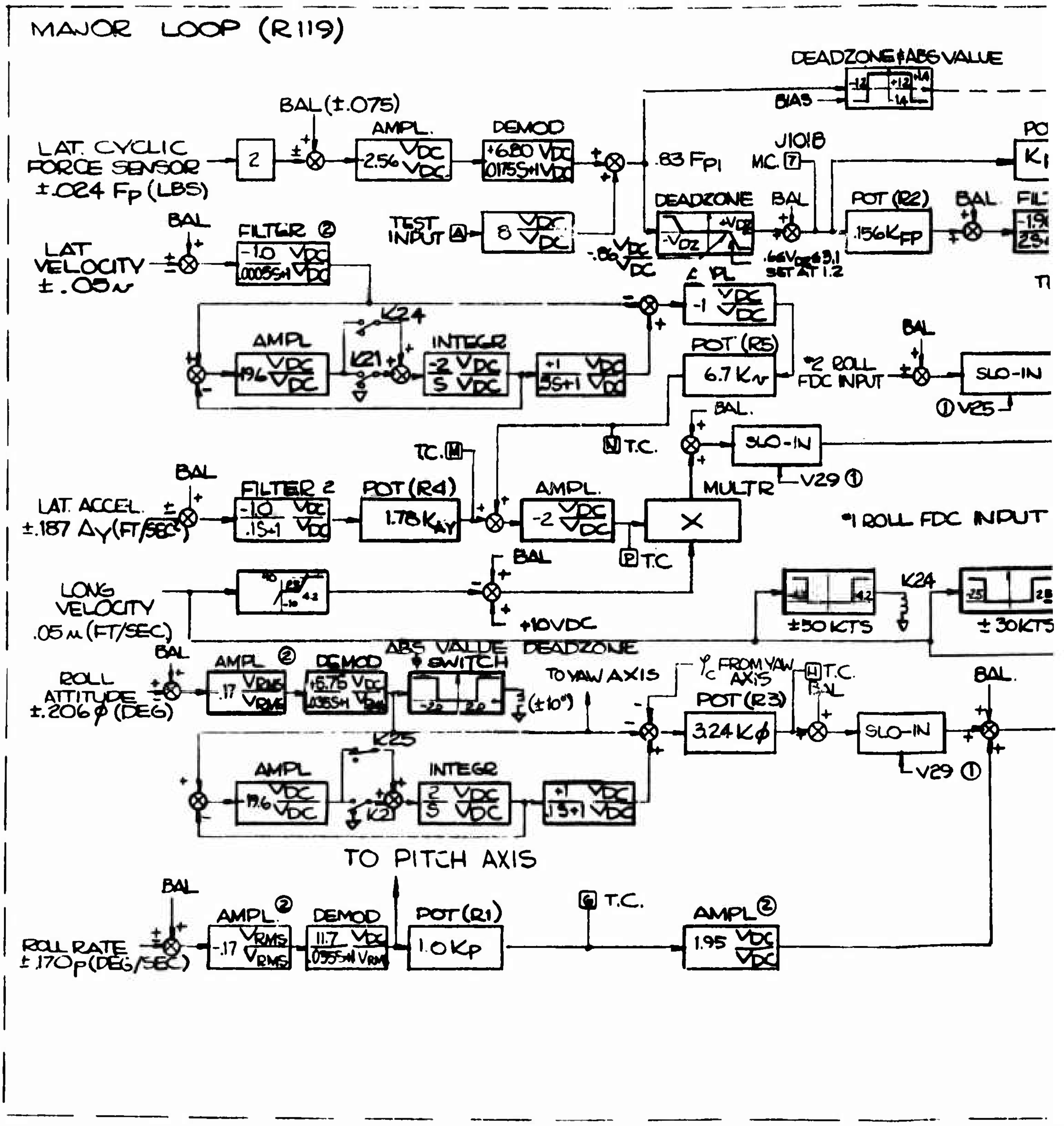
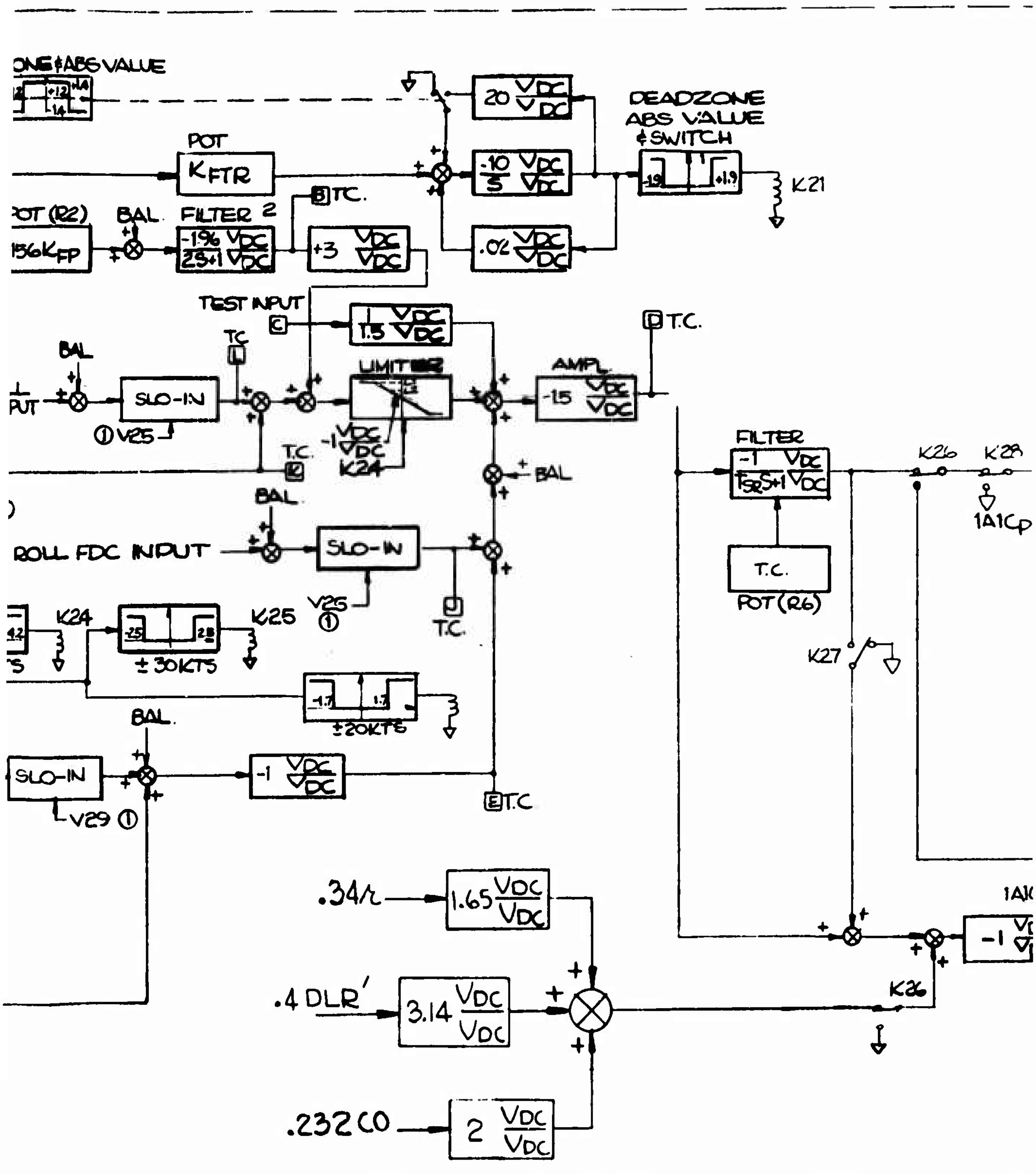
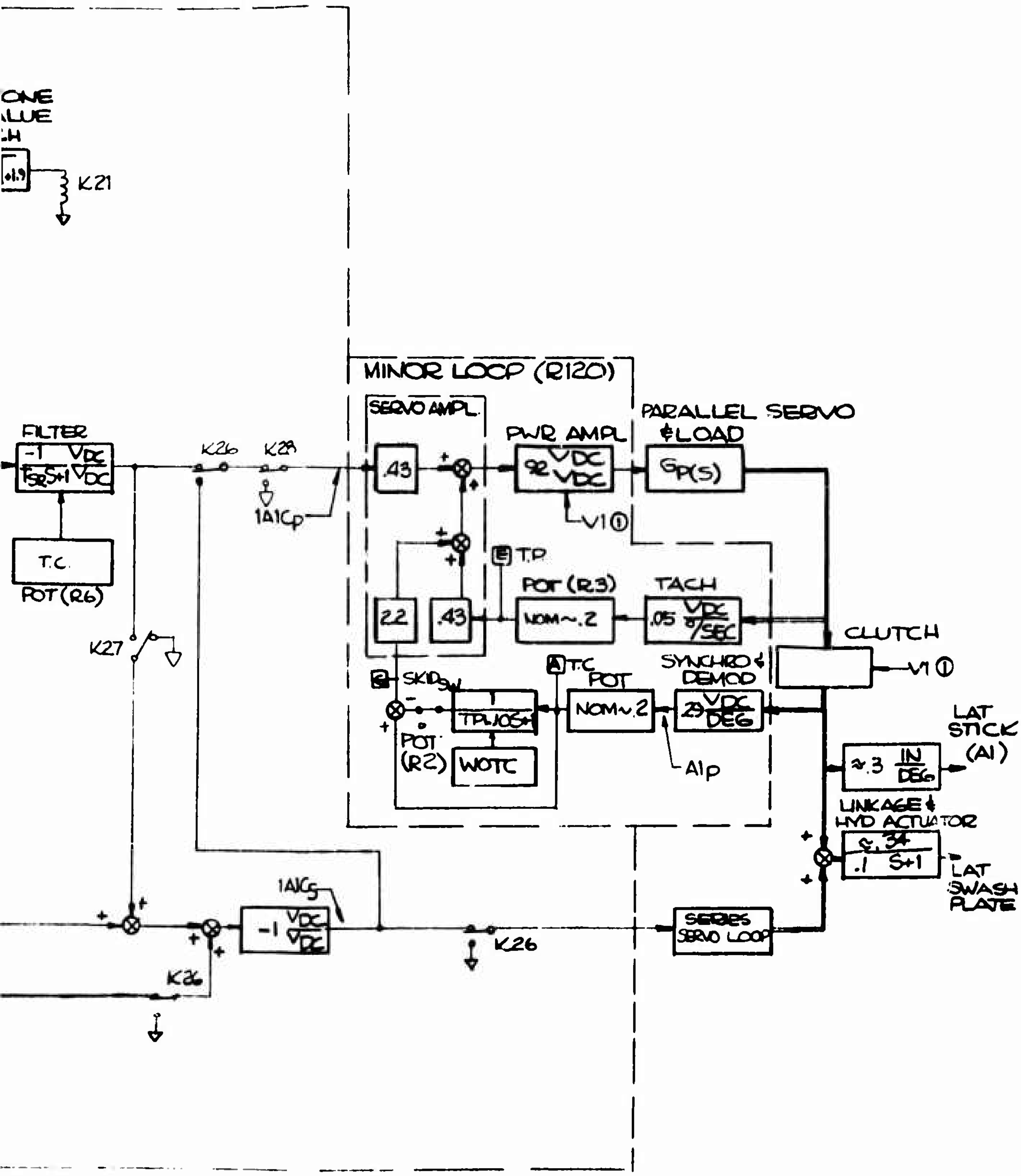


Figure 5. Roll Axis Block Diagram.


DEADZONE
ABS VALUE
& SWITCH
K21
POT
KFTR
BAL.
FILTER
TEST INPUT
T.C.
LIMITER
K24
AMPL.
-1.5
FILTER
T.C.
POT (R6)
K26
K28
1A1Cp
K27
BAL
ROLL FDC INPUT
SLO-IN
V25
K25
±30KTS
±20KTS
V29
.34r
1.65
.4 DLR'
3.14
.232 CO
2
VDC
VDC

B


ONE
LUE
H
K21
MINOR LOOP (R120)
SERVO AMPL.
PWR AMPL
PARALLEL SERVO
& LOAD
FILTER
K26
K28
.43
92 VDC/VDC
Gp(S)
1A1Cp
V1 ①
T.C.
POT (R6)
T.P.
POT (R3)
TACH
2.2
.43
NOM~.2
.05 VDC/°/SEC
CLUTCH
K27
SKID SW
T.C.
POT
SYNCHRO & DEMOD
NOM~.2
.29 VDC/DEG
POT (R2)
WOTC
A1p
≈.3 IN/DEG
LAT STICK (A1)
LINKAGE & HYD ACTUATOR
LAT SWASH PLATE
1A1CS
-1 VDC/VDC
K26
SERIES SERVO LOOP

1. "ENGAGE" - "OFF" (Roll Axis Engage)

2. "FDC" - "PAS"

For "normal" operation the above mode selector switches are set for "ENGAGE" and "PAS." For these switch settings, pilot force applied laterally to the cyclic stick results in the following:

1. Low Forward Speed Control (Hover and Slow Taxi) - For forward and rearward airspeeds below approximately 30 knots, a lateral force input commands a proportional incremental change in aircraft lateral airspeed. Upon release of the force input the system will attain a steady-state lateral airspeed, which is the aircraft airspeed that existed at the time the force was released. By applying small and rapid lateral force inputs, the pilot can maneuver the aircraft laterally in space without commanding a new trim lateral airspeed. The following PAS signals are used (see Figure 5):

   a. Roll rate feedback - for roll response shaping.

   b. Synchronized lateral airspeed - to provide an airspeed error signal. The airspeed error signal is forced to approximately zero in steady state by virtue of the closed-loop integrating action of the servo loop.

   c. Lateral acceleration - to provide lateral airspeed damping.

   d. Pilot electrical force input - to provide a flexible command input into the system.

2. Transition Control (Fast Taxi) - For forward speeds between approximately 30 and 50 knots, a lateral force input commands a blend of lateral airspeed and roll attitude. The PAS signals that are used are as follows:

   a. Roll rate feedback - for roll response shaping.

   b. Scheduled (as a function of forward airspeed) sum of synchronized lateral airspeed and lateral acceleration - to provide diminishing airspeed control, i.e., long-term control, as the aircraft transitions from hover to cruise.

c. Synchronized roll attitude - to provide an attitude error signal that is used as a short-term maneuvering reference.

d. Pilot electrical force input - command input.

3. Cruise Control - For forward airspeeds greater than 50 knots, a lateral force input commands a proportional incremental change in aircraft roll attitude. Upon release of the force input, the system will attain a steady-state roll attitude, which is the aircraft attitude that existed at the time the force was released. Small lateral translations can be made without disturbing the reference roll attitude by applying small and rapid lateral force inputs. The operation in this mode is essentially the same as that described for the pitch attitude hold mode.

Roll Axis Mode Selector Switching

The PAS is designed to minimize switching transients that occur when mode selector switches are thrown. The following transient suppression actions take place when typical mode selector switches are thrown:

1. Roll Axis Engage - From "OFF" to "ENGAGE."

   "FDC" - "PAS" switch - In "FDC" the FDC inputs are slowed in; in "PAS" the velocity and attitude inputs are slowed in.

2. Roll Axis Engage - From "ENGAGE" to "OFF." No transients occur (except due to feedback forces) since the servo is declutched.

3. "FDC" - "PAS" switch from "FDC" to "PAS."

   a. For "ENGAGE" setting - FDC input faded out and attitude and velocity signals faded in.

   b. For "OFF" - No transients.

4. "FDC" - "PAS" switch from "PAS" to "FDC."

   For "ENGAGE" setting - Airspeed and attitude inputs faded out and FDC input faded in.

Roll Axis Parameter Changes

The following roll axis parameters are adjustable via:

1. PAS Major Loop Electronics
   a. Roll rate gain (R1)
   b. Synchronized roll attitude gain (R3)
   c. Lateral acceleration gain (R4)
   d. Synchronized lateral airspeed gain (R5)
   e. Roll force gain (R2)
2. Minor Loop Electronics
   a. Tachometer feedback gain (R3)
   b. Follow-up washout time constant (R2)

SYSTEM MATH MODEL

A complete mathematical representation of the UH-1B system consists of descriptions of the following subsystems:

1. Vehicle (equations of motion)
2. Mechanical control system (math model)
3. Major loop computer (axis block diagrams)
4. Minor loop (block diagram)
5. Sensor characteristics

The math models of the above subsystems are described in the following sections.

Vehicle Math Model

USAAVLABS Furnished UH-1B Data

Data were furnished by USAAVLABS to describe the UH-1B. These data fall into the following general classifications:

1. Aircraft Stability Derivatives
2. Aircraft Time Responses
3. Aircraft Trim Conditions
4. Aircraft Mechanical Drawings
5. Aircraft Operation and Maintenance Manuals

Most of the above data, which are pertinent to a first-cut development of the system math model, have been reviewed, as required. The following comments apply to the data:

1. Aircraft Stability Derivatives - the data format is excellent with respect to defining a math model of the vehicle; the accuracy of the data, i.e., how well the parameters represent the actual aircraft parameters, was assumed to be good (pending the results of future checks with an instrumented aircraft).

2. Aircraft Time Responses - the data format is excellent with respect to providing a check for ANC's digital simulation of the aircraft; here again, the data were assumed to be accurate until this could be verified with an instrumented aircraft.

3. Aircraft Trim Conditions - same comments as 1 and 2 above.

4. Aircraft Mechanical Drawings - the drawings have been reviewed with respect to arriving at approximate locations of the servos.

5. Aircraft Operation and Maintenance Manuals - the manuals have been reviewed with respect to arriving at approximate locations of the servos and to generate first-cut math models of the control systems.

Math Model of Vehicle Aerodynamics

The math model of the UH-1B for any given set of steady-state operating conditions is described by the Vell C-81

printouts. Each printout is titled "Bell Helicopter IBM 360/Program AGAJ68/Helicopter Rigid Body Dynamics Analysis/ Compiled 12/13/68."

The Bell program (C-81) is input with basic aircraft data and generates the following:

1. Trim conditions.

2. Control effectiveness terms entitled "Partial Derivative Matrix."

3. Stability derivative terms entitled "Stability Partial Derivative Matrix."

4. Longitudinal three-degree-of-freedom equations of motion.

5. The characteristic roots of the longitudinal three-degree-of-freedom equations.

6. Lateral three-degree-of-freedom equations of motion.

7. The characteristic roots of the lateral three-degree-of-freedom equations.

8. Time domain printouts and print plots of the system (six-degree-of-freedom) response to control inputs.

A summary of forward flight trim conditions (taken from digital runs received on 2/4/70) is given in Figure 6. Figure 6 shows the following:

1. Roll and pitch attitude vs airspeed.

2. Main and tail rotor power vs airspeed.

3. Control surface position vs airspeed.

Table IV describes digital program outputs (partial and stability derivative matrices) that were used to:

1. Deduce the lateral and longitudinal three-degree-of-freedom equations of motion (see Table V) used in the digital program.

2. Construct a six-degree-of-freedom model of the aircraft (see Table VI).

The lateral and longitudinal three-degree-of-freedom equations of motion parameters for each flight condition were

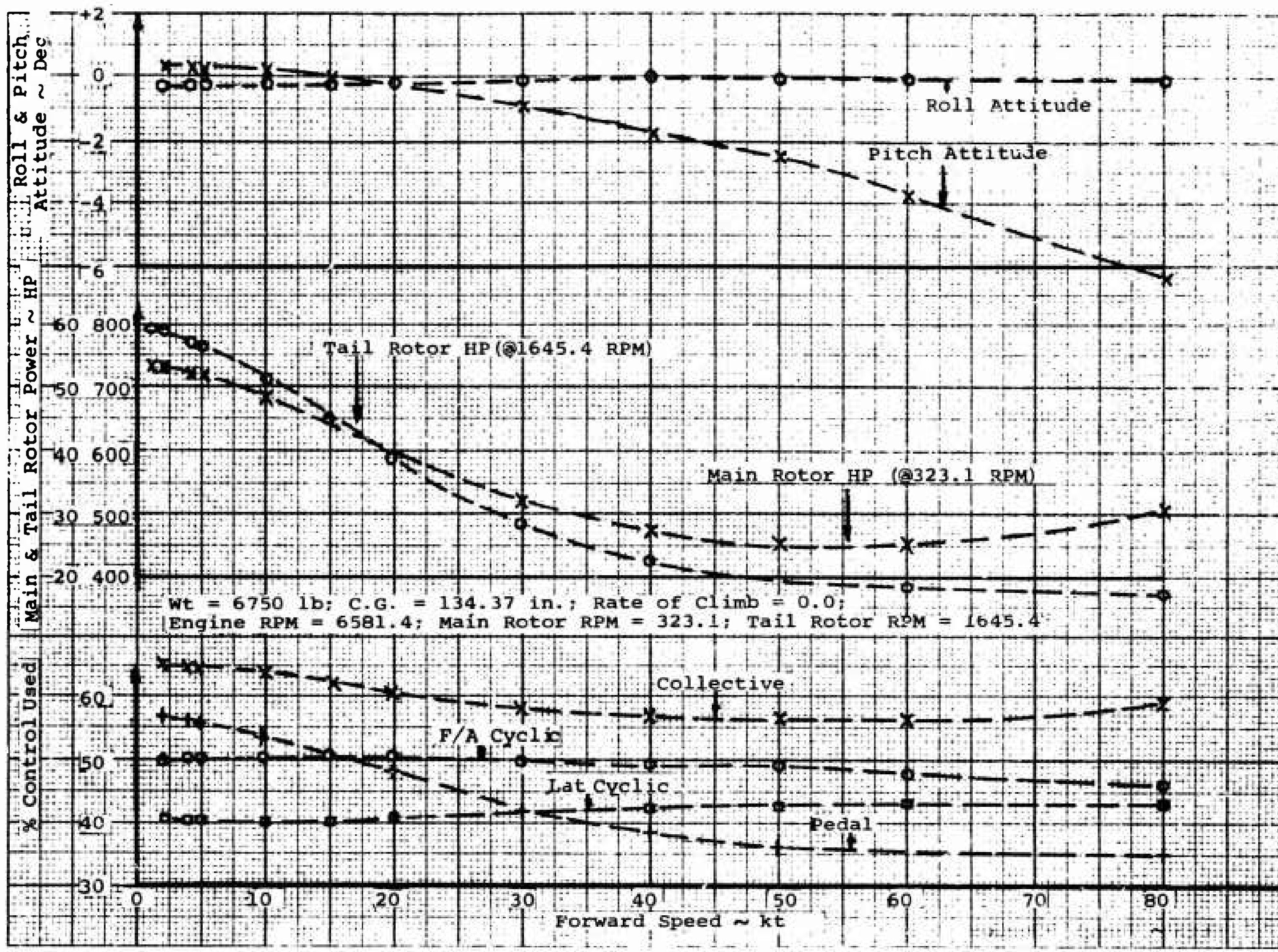


Figure 6. UH-1B Forward Flight Trim Conditions From C-81 Runs.

TABLE IV. C-81 PARTIAL AND STABILITY DERIVATIVE MATRICES

<u>Partial Derivative Matrix</u>

	X (lb)	Y (lb)	Z (lb)	N (f_p)*	M (f_p)	L (f_p)
Coll (in)	$X_{CO}\left(\frac{lb}{in}\right)$	$Y_{CO}\left(\frac{lb}{in}\right)$	$Z_{CO}\left(\frac{lb}{in}\right)$	$N_{CO}\left(\frac{f_p}{in}\right)$	$M_{CO}\left(\frac{f_p}{in}\right)$	$L_{CO}\left(\frac{f_p}{in}\right)$
F/A Cyc (in)	$X_{B1}\left(\frac{lb}{in}\right)$	$Y_{B1}\left(\frac{lb}{in}\right)$	$Z_{B1}\left(\frac{lb}{in}\right)$	$N_{B1}\left(\frac{f_p}{in}\right)$	$M_{B1}\left(\frac{f_p}{in}\right)$	$L_{B1}\left(\frac{f_p}{in}\right)$
Lat Cyc (in)	$X_{A1}\left(\frac{lb}{in}\right)$	$Y_{A1}\left(\frac{lb}{in}\right)$	$Z_{A1}\left(\frac{lb}{in}\right)$	$N_{A1}\left(\frac{f_p}{in}\right)$	$M_{A1}\left(\frac{f_p}{in}\right)$	$L_{A1}\left(\frac{f_p}{in}\right)$
Pedal (in)	$X_{DLR}\left(\frac{lb}{in}\right)$	$Y_{DLR}\left(\frac{lb}{in}\right)$	$Z_{DLR}\left(\frac{lb}{in}\right)$	$N_{DLR}\left(\frac{f_p}{in}\right)$	$M_{DLR}\left(\frac{f_p}{in}\right)$	$L_{DLR}\left(\frac{f_p}{in}\right)$

*foot-pounds

TABLE IV - Continued

Stability Partial Derivative Matrix

	X (lb)	Y (lb)	Z (lb)	N (f_p)*	M (f_p)	L (f_p)
p (rad/sec)	X_p (lb-sec)	Y_p (lb-sec)	Z_p (lb-sec)	N_p (f_p-sec)	M_p (f_p-sec)	L_p (f_p-sec)
q (rad/sec)	X_q (lb-sec)	Y_q (lb-sec)	Z_q (lb-sec)	N_q (f_p-sec)	M_q (f_p-sec)	L_q (f_p-sec)
r (rad/sec)	X_r (lb-sec)	Y_r (lb-sec)	Z_r (lb-sec)	N_r (f_p-sec)	M_r (f_p-sec)	L_r (f_p-sec)
u (ft/sec)	$X_u \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	$Y_u \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	$Z_u \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	N_u (lb-sec)	M_u (lb-sec)	L_u (lb-sec)
v (ft/sec)	$X_v \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	$Y_v \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	$Z_v \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	N_v (lb-sec)	M_v (lb-sec)	L_v (lb-sec)
w (ft/sec)	$X_w \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	$Y_w \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	$Z_w \left(\frac{\text{lb-sec}}{\text{ft}}\right)$	N_w (lb-sec)	M_w (lb-sec)	L_w (lb-sec)

*foot-pounds

TABLE V. C-81 3-DEGREE-OF-FREEDOM EQUATIONS OF MOTION

<u>Longitudinal Equations of Motion</u>

$$\begin{bmatrix} \frac{W}{G}S - X_u & -X_w & \left(-\frac{X_q}{U_o}+\frac{W}{GU_o}\right)S+\frac{W}{U_o} \\ -Z_u & \frac{W}{G}S - Z_w & -\left(\frac{W}{G}+\frac{Z_q}{U_o}\right)S+\frac{W}{U_o}\sin\theta_o \\ -M_u & -M_w & \frac{I_Y}{U_o}S^2 - \frac{M_q}{U_o}S \end{bmatrix} \begin{bmatrix} U\left(\frac{ft/sec}{ft/sec}\right) \\ \alpha\ (rad) \\ \theta\ (rad) \end{bmatrix} = \begin{bmatrix} \frac{X_{B1}}{U_o} & \frac{X_{CO}}{U_o} \\ \frac{Z_{B1}}{U_o} & \frac{Z_{CO}}{U_o} \\ \frac{M_{B1}}{U_o} & \frac{M_{CO}}{U_o} \end{bmatrix} \begin{bmatrix} B1\ (in) \\ CO\ (in) \end{bmatrix}$$

<u>Lateral Equations of Motion</u>

$$\begin{bmatrix} \frac{W}{G}S - Y_v & -\left(\frac{Y_p}{U_o}+\frac{W}{G}\frac{W_o}{U_o}\right)S-\frac{W}{U_o} & -\frac{Y_r}{U_o} \\ -L_v & \frac{I_X}{U_o}S^2 - \frac{L_p}{U_o}S & -\frac{I_{XZ}}{U_o}S-\frac{L_r}{U_o} \\ -N_v & -\frac{I_{XZ}}{U_o}S^2-\frac{N_p}{U_o}S & \frac{I_Z}{U_o}S-\frac{N_r}{U_o} \end{bmatrix} \begin{bmatrix} \beta\ (rad) \\ \varphi\ (rad) \\ r\ (rad/sec) \end{bmatrix} = \begin{bmatrix} \frac{Y_{A1}}{U_o} & \frac{Y_{DLR}}{U_o} \\ \frac{L_{A1}}{U_o} & \frac{L_{DLR}}{U_o} \\ \frac{N_{A1}}{U_o} & \frac{N_{DLR}}{U_o} \end{bmatrix} \begin{bmatrix} A1\ (in) \\ DLR\ (in) \end{bmatrix}$$

TABLE VI. UH-1B 6-DEGREE-OF-FREEDOM EQUATIONS OF MOTION

$$
\begin{bmatrix}
\frac{W}{G}S-X_u & -X_v & -X_w & -\frac{X_p}{U_o}S & \left(-\frac{X_q}{U_o}+\frac{W}{GU_o}\right)S+\frac{W}{U_o} & -\frac{X_r}{U_o} \\
-Y_u & \frac{W}{G}S-Y_v & -Y_w & \left(-\frac{Y_p}{U_o}+\frac{W}{G}\frac{W_o}{U_o}\right)S-\frac{W}{U_o} & -\frac{Y_q}{U_o}S & -\frac{Y_r}{U_o} \\
-Z_u & -Z_v & \frac{W}{G}S-Z_w & -\frac{Z_p}{U_o}S & -\left(\frac{W}{G}+\frac{Z_q}{U_o}\right)S+\frac{W}{U_o}\sin\theta_o & -\frac{Z_r}{U_o} \\
-L_u & -L_v & -L_w & \frac{I_X}{U_o}S^2-\frac{L_p}{U_o}S & -\frac{L_q}{U_o}S & -\frac{I_{XZ}}{U_o}S-\frac{L_r}{U_o} \\
-M_u & -M_v & -M_w & -\frac{M_p}{U_o}S & \frac{I_Y}{U_o}S^2-\frac{M_q}{U_o}S & -\frac{M_r}{U_o} \\
-N_u & -N_v & -N_w & -\frac{I_{XZ}}{U_o}S^2-\frac{N_p}{U_o}S & -\frac{N_q}{U_o}S & \frac{I_Z}{U_o}S-\frac{N_r}{U_o}
\end{bmatrix}
\begin{bmatrix}
U\ \frac{ft/sec}{ft/sec} \\
\beta\ (rad) \\
\alpha\ (rad) \\
\varphi\ (rad) \\
\theta\ (rad) \\
r\ (rad/sec)
\end{bmatrix}
=
$$

TABLE VI – Continued

$$\begin{bmatrix} \frac{X_{B1}}{U_o} & \frac{X_{CO}}{U_o} & \frac{X_{A1}}{U_o} & \frac{X_{DLR}}{U_o} \\ \frac{Y_{B1}}{U_o} & \frac{Y_{CO}}{U_o} & \frac{Y_{A1}}{U_o} & \frac{Y_{DLR}}{U_o} \\ \frac{Z_{B1}}{U_o} & \frac{Z_{CO}}{U_o} & \frac{Z_{A1}}{U_o} & \frac{Z_{DLR}}{U_o} \\ \frac{L_{B1}}{U_o} & \frac{L_{CO}}{U_o} & \frac{L_{A1}}{U_o} & \frac{L_{DLR}}{U_o} \\ \frac{M_{B1}}{U_o} & \frac{M_{CO}}{U_o} & \frac{M_{A1}}{U_o} & \frac{M_{DLR}}{U_o} \\ \frac{N_{B1}}{U_o} & \frac{N_{CC}}{U_o} & \frac{N_{A1}}{U_o} & \frac{N_{DLR}}{U_o} \end{bmatrix} \begin{bmatrix} B1 \ (in.) \\ CO \ (in.) \\ A1 \ (in.) \\ DLR \ (in.) \end{bmatrix}$$

inserted in ANC's root cracking program. The resulting characteristic equations' roots correlated with the roots given in USAAVLABS digital computer runs.

Figure 7 shows a comparison of stability derivatives between two sets of data: (1) C-81 data for the standard UH-1B without stabilizer bar; and (2) data for a "normal" single-rotor helicopter (see reference 5). Comparing the C-81 data with the UH-1B data we can see that:

1. The rotary damping derivatives from the C-81 data, M_q and L_p, are quite low (especially M_q) at low speed and undergo a fairly large change with speed.

2. The speed stability derivatives from the C-81 data, M_u and L_v, are pretty much unlike their counterparts. M_u is positive in hover but has the characteristics of a tandem (i.e., becomes negative after transition) in forward flight. L_v shows a reversal tendency around 40 kt. Both derivatives undergo a fairly large change with speed.

3. The angle-of-attack derivative from the C-81 data, U_oM_w, is positive and appears to vary as a higher order of speed than first order. This derivative also seems to display tandem rotor characteristics in forward flight.

M_u, which with L_v determines the aircraft's gust sensitivity about the pitch and roll axes, and U_oM_w appear to be the wrong sign in forward flight. This contributes to a characteristic root that is fairly far out in the right-half plane in forward flight. Response to a pulse input in pitch at 80 kt yields a pitch rate time history that does not show any significant reversal tendencies (like one might expect from the actual aircraft).

Figures 8, 9 and 10 show C-81 digital time domain responses that have been plotted. This type of response data was used to check ANC's digital time response; i.e., CSMP simulation.

A large portion of the analysis work was conducted using three level, forward-flight (2, 40 and 80 kt) conditions. This was done so as not to detract from the basic objective (i.e., to set the stage for further work) by limiting its scope and also to minimize any analytical work which might be rendered useless by some lack of model correlation (no significant flight test data was available) with the real world (e.g., using a vehicle with different rotor characteristics, aircraft model inaccuracies, not predicting some installed sensor idiosyncrasies, inaccurate mechanical control system characteristics, etc.).

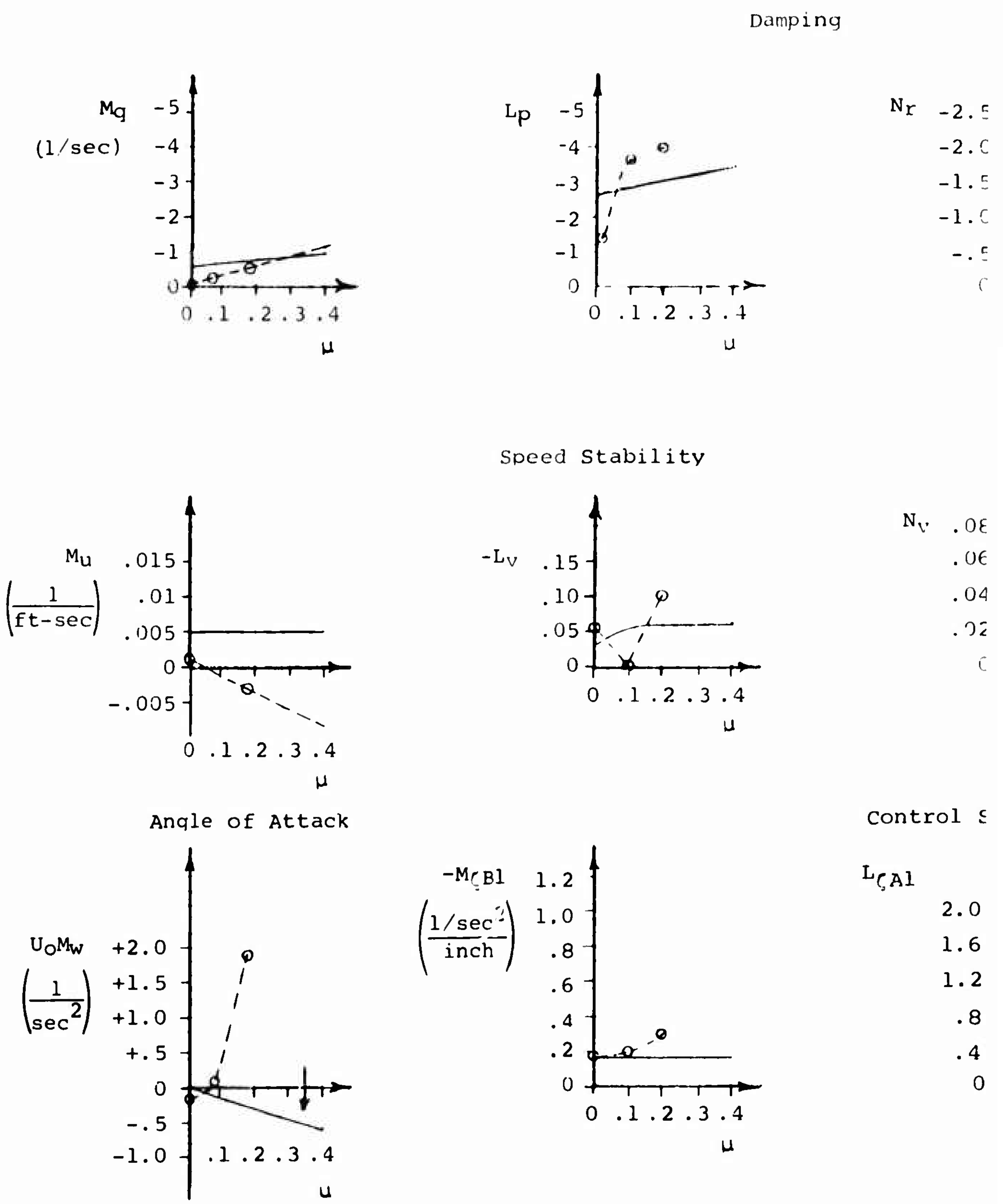
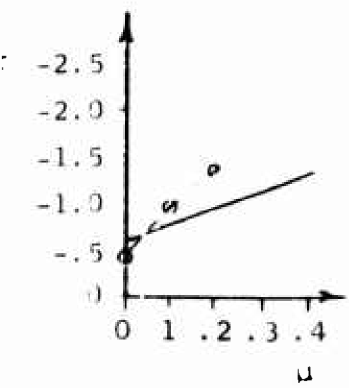
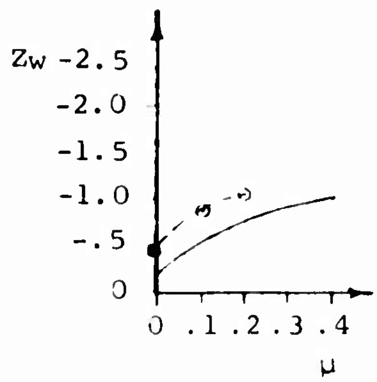
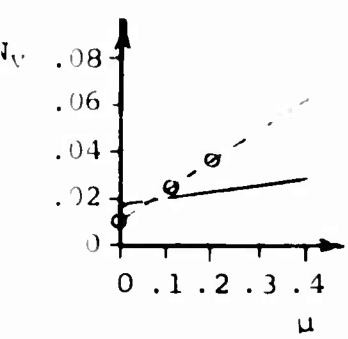


Figure 7. Stability Derivative Comparison.

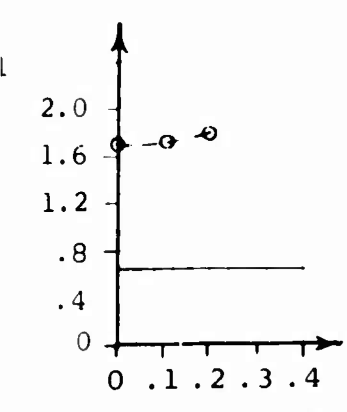
A

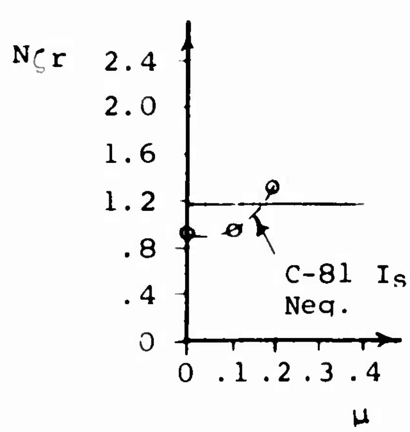






trol Sensitivity





NOTES

1. μ = Tip Speed Ratio = $U_0/\Omega R$;

UH-1B = ΩR = (324)(22)(π/30)

= 748 ft/sec

2. Symbols

⊙ ~ UH-1B (C-81 Body Axis Data)

— ~ "Normal" single rotor value (Reference 5)

B

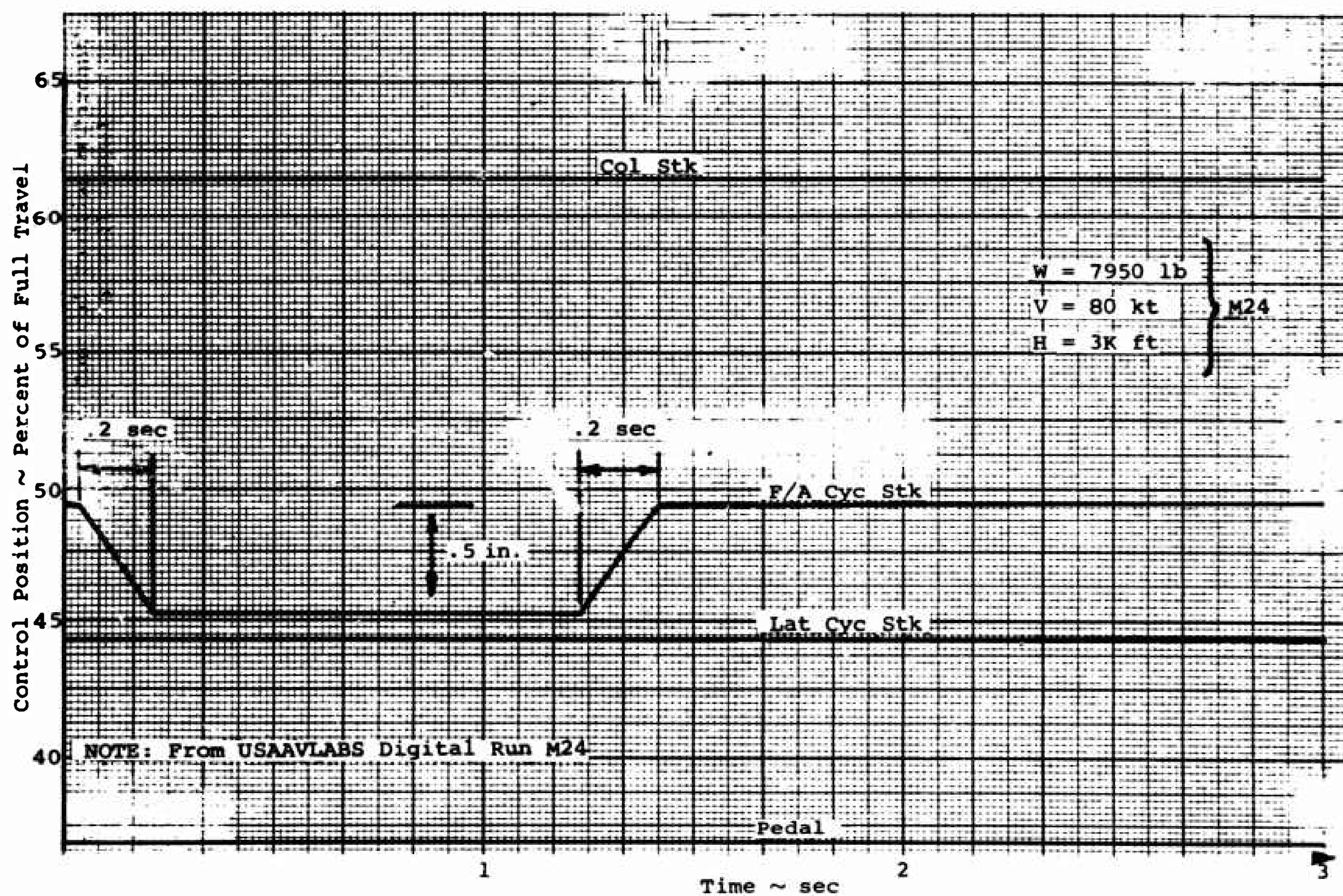


Figure 8. C-81 Control Position vs Time.

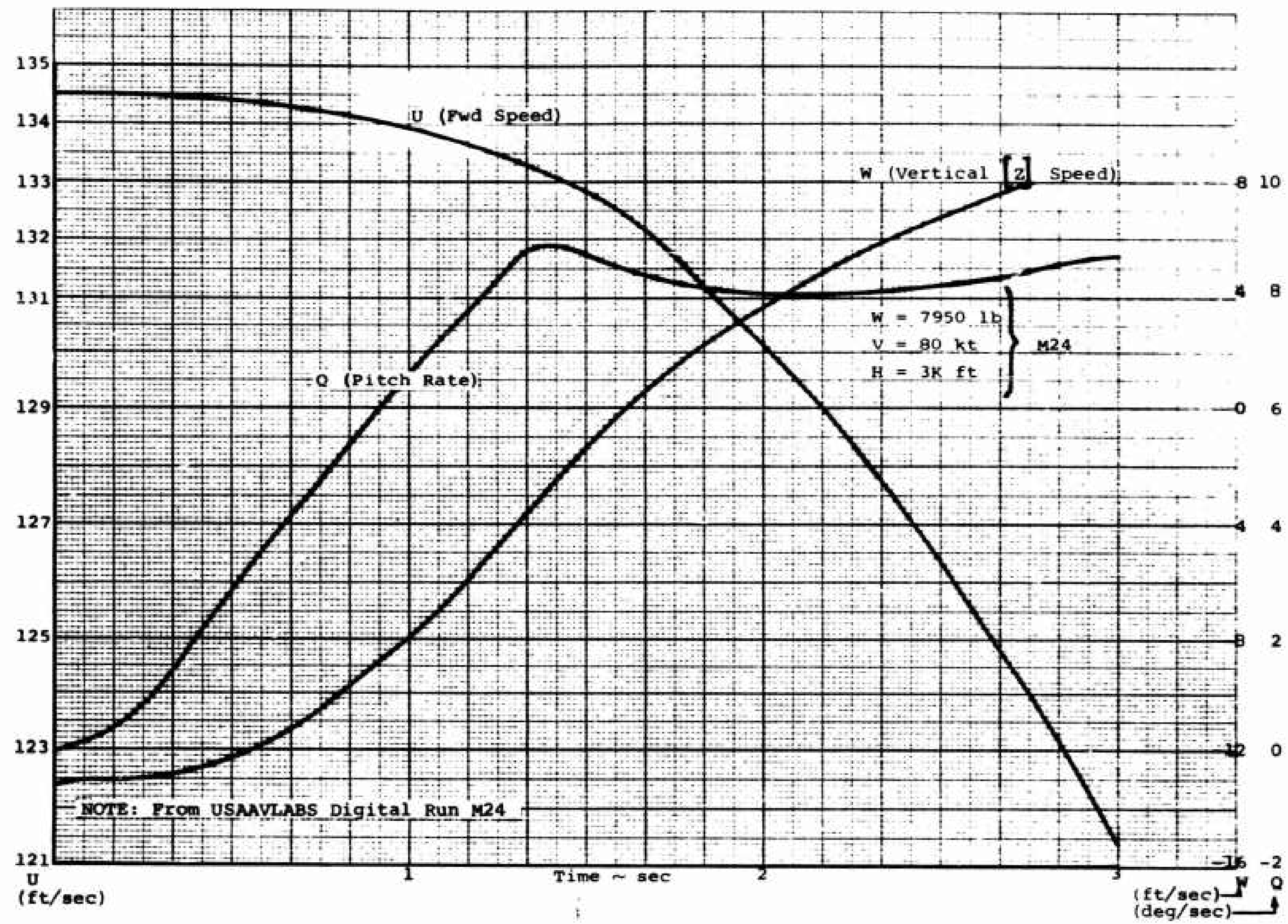


Figure 9. C-81 Longitudinal Parameters vs Time.

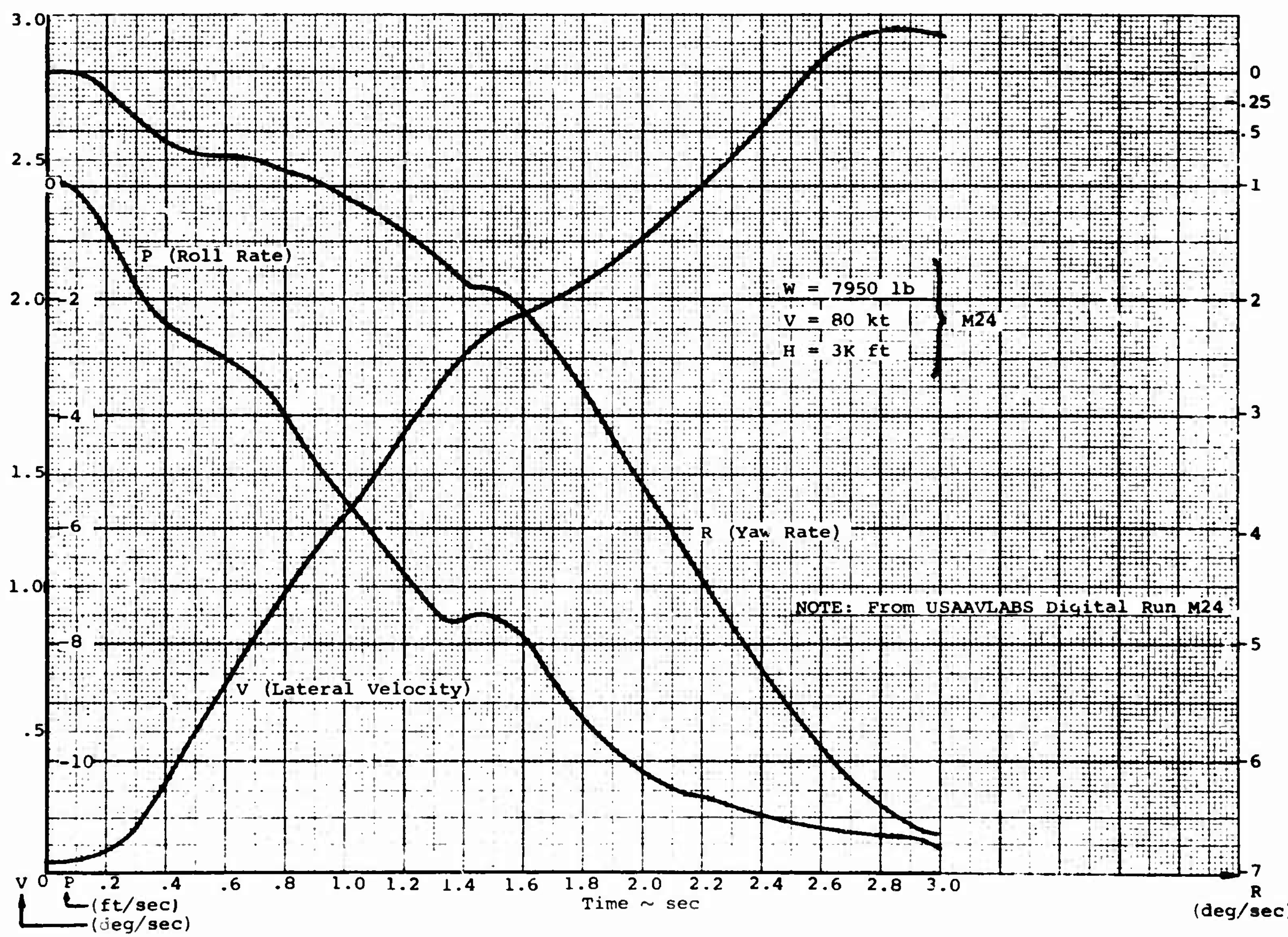


Figure 10. C-81 Lateral Parameters vs Time.

Approximate Location of Force Sensors

The force sensors will be located either within the grip (and pedal) or on the first linkage downstream. Present preference is for the former approach.

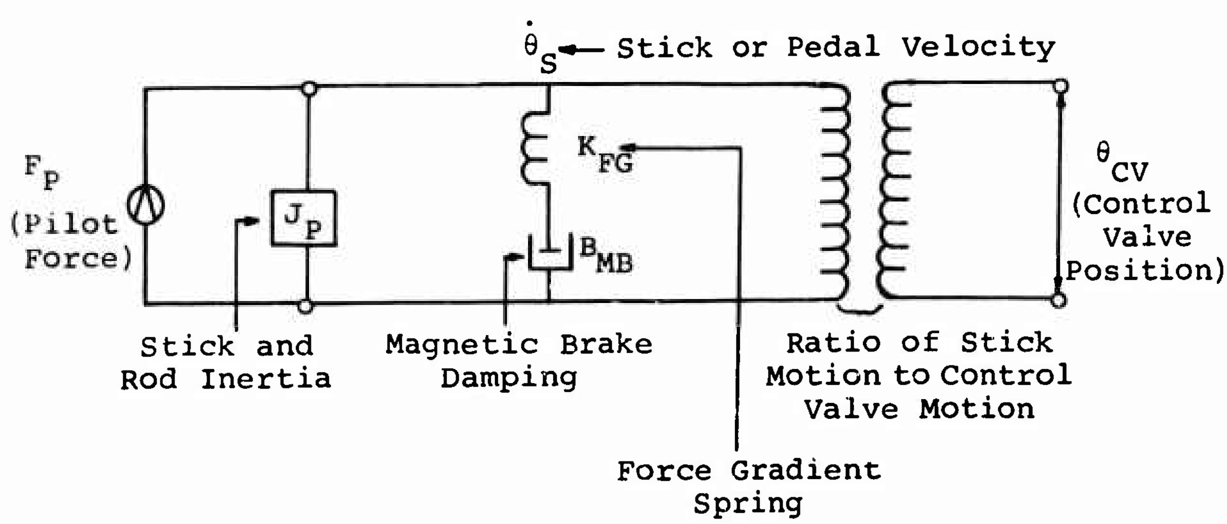
Approximate Location of Servos

From examination of an actual vehicle and vehicle drawings it appears that the servos; i.e., longitudinal cyclic, lateral cyclic, collective and tail rotor, should be located somewhere between fuselage locations F.S. 70 and F.S. 120. The selected area is located beneath the cabin floor between F.S. 70 and F.S. 120 (approximately by the ends of the cargo door).

UH-1B Mechanical Control System Math Models

Lateral Cyclic, Longitudinal Cyclic and Pedal

The lateral cyclic, longitudinal cyclic and pedal mechanical control systems can be represented by the following electrical analog:



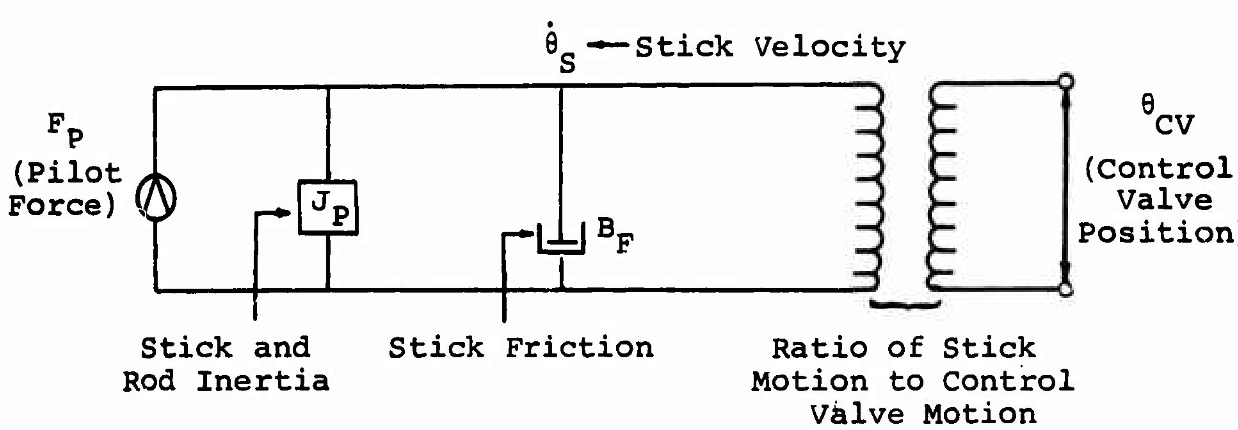
Assuming that the control valve does not reflect any significant load to the stick, we have

$$F_P = \theta_S \left(J_P\, S^2 + \frac{B_{MB}\, K_{FG}\, S}{B_{MB}\, S + K_{FG}} \right) = \theta_S \left[\frac{J_P\, S^2 \left(B_{MB}\, S + K_{FG} \right) + B_{MB}\, K_{FG}\, S}{B_{MB}\, S + K_{FG}} \right]$$

$$\frac{S\theta_S}{F_P} = \frac{B_{MB}\, S + K_{FG}}{J_P\, B_{MB}\, S^2 + K_{FG}\, J_P\, S + B_{MB}\, K_{FG}} = \left(\frac{1}{B_{MB}} \right) \left(\frac{\frac{B_{MB}}{K_{FG}}\, S + 1}{\frac{J_P}{K_{FG}}\, S^2 + \frac{J_P}{B_{MB}}\, S + 1} \right)$$

Collective

The collective mechanical control system can be represented by



From the above figure,

$$F_P = \theta_S \left(J_P\, S^2 + B_F\, S \right); \quad \frac{S\theta_S}{F_P} = \frac{1}{B_F} \left(\frac{1}{\frac{J_P}{B_F}\, S + 1} \right)$$

Major Loop Computer

The major loop computer encompasses the PAS electronics between the aircraft sensor and pilot force sensor inputs and the servo command signals in each of the four control axes. The major loop computer (R119) math model is defined in Figures 1, 3, 4 and 5. These figures are a near one-to-one (i.e., the blocks for the most part do not represent lumping circuits) representation of the major loop electronics.

Figures 1, 3, 4 and 5 represent a math model of the major loop computer in the following respects:

1. The dynamics, scaling and polarities (except for the input signal polarity which is left flexible to accommodate either sign of signal input) for each leg are obvious by inspection of the figures.

2. The internally initiated switching and the effects of this switching are shown.

Figure 1 is a block diagram of the pitch axis of the PAS.

The pitch force input signal (from the longitudinal cyclic force sensor) is first amplified to increase the signal level and then splits into the following paths:

1. A control input signal path through a force breakout, force gradient pot, force input shaping filter and some amplification.

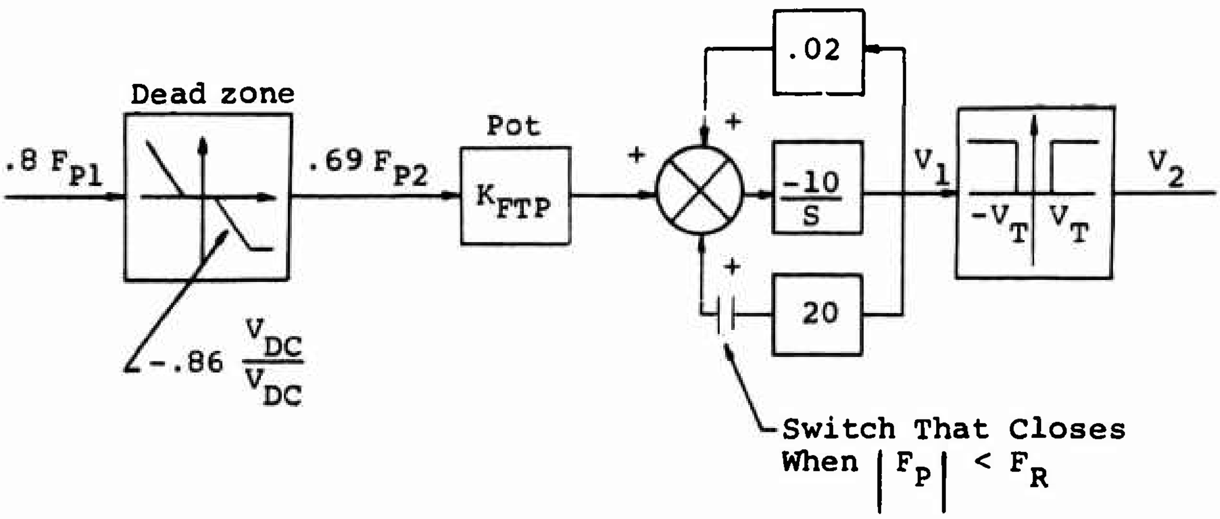
2. A force trigger path which initiates either longitudinal velocity of pitch attitude synchronization when the force dead-zone is exceeded for a certain period of time.

3. A force trigger reset path which causes rapid reset of the force trigger relay, K1, when the force input is reduced below the dead-zone value.

The pitch cyclic force trigger path logic is mechanized to yield a switching time that is approximately inversely proportional to the incremental force (pilots) above the electrical force dead-zone. Upon pilot release of force (or more exactly, reducing force to below a set level), the force logic is rapidly reset (with approximately a .005-second time constant) to zero. The rapid reset feature assures repeatable logic by virtue of starting from the same initial

conditions on successive applications of pilot force.

The figure below is a block diagram representation of the pitch pilot force logic circuitry (roll is identical).



From the block diagram, we have

$$V_1 = 34.5\ K_{FTP} \frac{F_{P2}}{5S + 1} \tag{1}$$

For fairly rapid application of force (F_{P2} is approximately a step function of magnitude F_{P2}), we get for equation (1)

$$v_1(t) = 34.5\ K_{FTP}F_{P2}(1 - \epsilon^{-t/5}) \tag{2}$$

In the linear region (≈20% of steady state value) of $v_1(t)$, we have

$$v_1(t) \approx \frac{34.5\ K_{FTP}F_{P2}\ t}{5} = 6.9\ K_{FTP}F_{P2}\ t \tag{3}$$

We see that V_2 will switch for

$$\left|6.9\ K_{FTP}F_{P2}\ t\right| = V_T \quad \text{or} \quad t = \frac{V_T}{6.9\ K_{FTP}F_{P2}} \tag{4}$$

For $$V_T = 1.9 \text{ volts}$$

$$K_{FTP} = 1.0$$

$$t = \frac{.276}{F_{P2}} \text{ second}$$

Where F_{P2} = excess force above breakout in pounds.

When the switch closes (i.e., for $|F_P| < F_R$) and F_P is below the electrical dead-zone, V_1 will reset to zero with a time constant (reciprocal of loop gain) of .005 second.

A test input path is summed into the force path (prior to the force breakout) for inserting simulated force input signals.

The longitudinal velocity path contains the following:

1. A sensor noise filter which presently contains a very short time constant (that can be changed quickly, if necessary).

2. A synchronizer circuit that either synchronizes to the existing velocity (with a time constant of 25 ms) or holds a previous value of velocity. A lag is included in one leg of the synchronizer to smooth the net velocity signal when the synchronizer circuit goes from synchronization to hold while the aircraft is changing velocity.

3. A velocity gain potentiometer and amplification circuit.

The longitudinal acceleration path contains a sensor noise filter and a gain potentiometer prior to summing with the longitudinal velocity leg.

The sum of longitudinal velocity and acceleration is amplified, fed through a slo-in (variable gain of 0 to -1 during engage), a pitch rate command limiter (during maneuvering), and an amplification circuit.

The output from the vertical gyro (pitch attitude) is demodulated and splits into the following control paths each of which has its own gain adjustment pot:

1. A smoothed attitude synchronizer/hold path that functions like the longitudinal velocity circuitry.

2. A washed-out attitude path that can be used in conjunction with the velocity loop to command short-term pitch attitude. The limiter would then be a short-term attitude command limiter.

The sum of synchronized and washed out attitude is fed through a slo-in to an amplification circuit.

The output from the pitch rate gyro is demodulated (the demodulator filter can serve as a noise filter, if required) and passed through a gain potentiometer and an amplification circuit.

The algebraic sum of the major loop signals is fed as servo position commands to both the parallel servo (through a low-pass filter) and the series servo (through a high-pass filter).

The math model of the other three axes of control (roll, yaw and collective) are defined in Figures 3, 4 and 5. Descriptions of the control action in these three axes are similar to the pitch axis, however, for simplicity these descriptions will be omitted.

Minor Loop Characteristics

General Description

American Nucleonics Corporation is using a dc servo motor which it has developed. The minor loop consists of a high-response, high-torque, lightweight dc torque motor servo and the associated electronics (minor loop computer).

The parallel servo motor assembly consists of a high-velocity, low-torque dc motor, followed by a gear ratio of approximately 100:1 to the output shaft. The gearbox contains a ball detent disengage clutch so that when the system is not in use but power is applied, the complete inertia of the motor reflected through the gearbox is not seen at the output shaft. A synchro is used as a position pickoff, and is connected directly to the output shaft. A tachometer, used for rate damping, is geared to the motor shaft with a 1:1 ratio.

As a separate parallel servo, the complete servo motor assembly weighs less than 3 pounds, is packaged in a volume of approximately 40 cubic inches, and has a minimum torque and speed capability of 200 inch-pounds and 3.0 radians/second, respectively.

The minor loop computer consists of a power amplifier capable of delivering up to 100 watts of power into the 5.5-ohm motor, a summing amplifier, a demodulator for the synchro feedback, a current limiter for the power amplifier, and the necessary power supplies. The current limiter is adjustable for each direction of the motor. This circuitry ensures that the phenomenon of motor "plugging" cannot occur.

The forward loop gain of the servo (power) amplifier is chosen to be relatively high so that a high bandwidth and resistance to force inputs at the motor output shaft are obtained.

The use of a position washout filter allows the motor to act as an automatic trim device to control inputs when the aircraft is in the air. On the ground, the motor acts as a position device; i.e., skid switches remove the washout.

Minor Loop Math Model

Figure 11 is a block diagram of the torque motor servo (described by the characteristics shown in Table VII) and the minor loop computer electronics which comprise the minor loop.

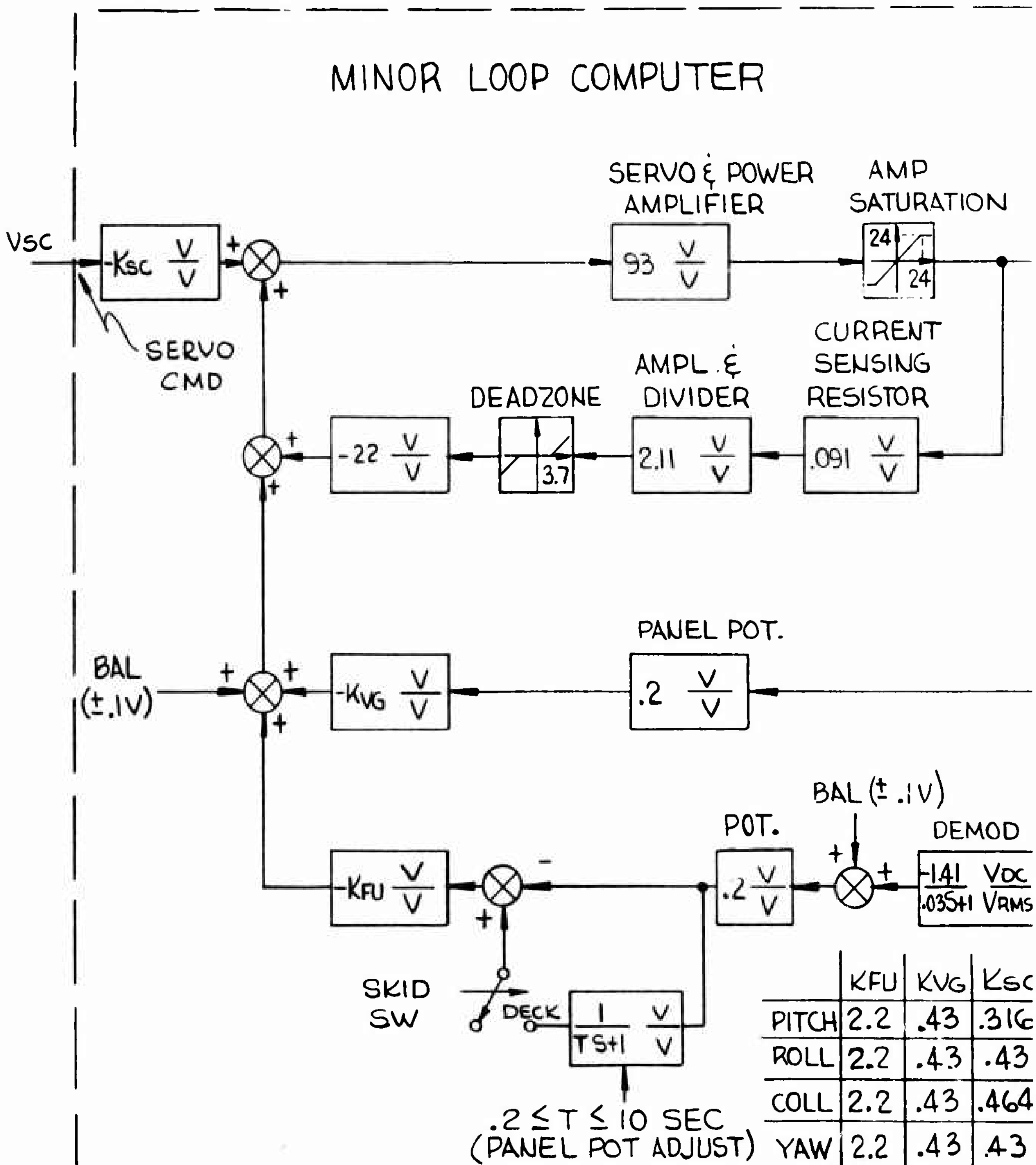
The motor coil has a nominal dc resistance of 5.5 ohms and a time constant of .0005 second. The torque versus current relationship is fairly linear out to approximately 2.5 inch-pounds (at motor output).

The motor inertia is probably larger than the reflected load inertia. Only the motor inertia is shown in Figure 11. The open-loop position loop characteristics are approximately 6000/S. Therefore, there is room for an increase in total inertia before a significant change in closed-loop characteristics occurs.

Motor damping (from torque-speed characteristics), breakaway torque, induced back EMF (BEMF), backlash and gearing from motor to output shaft are all described in Figure 11. Nominal tachometer and synchro (position) gradients complete the representation of the torque motor servo shown in Figure 11.

A ball-park feel spring rate, i.e., yielding approximately 1 pound of pilot force per inch of stick travel, has been included to represent this mechanical control system characteristic.

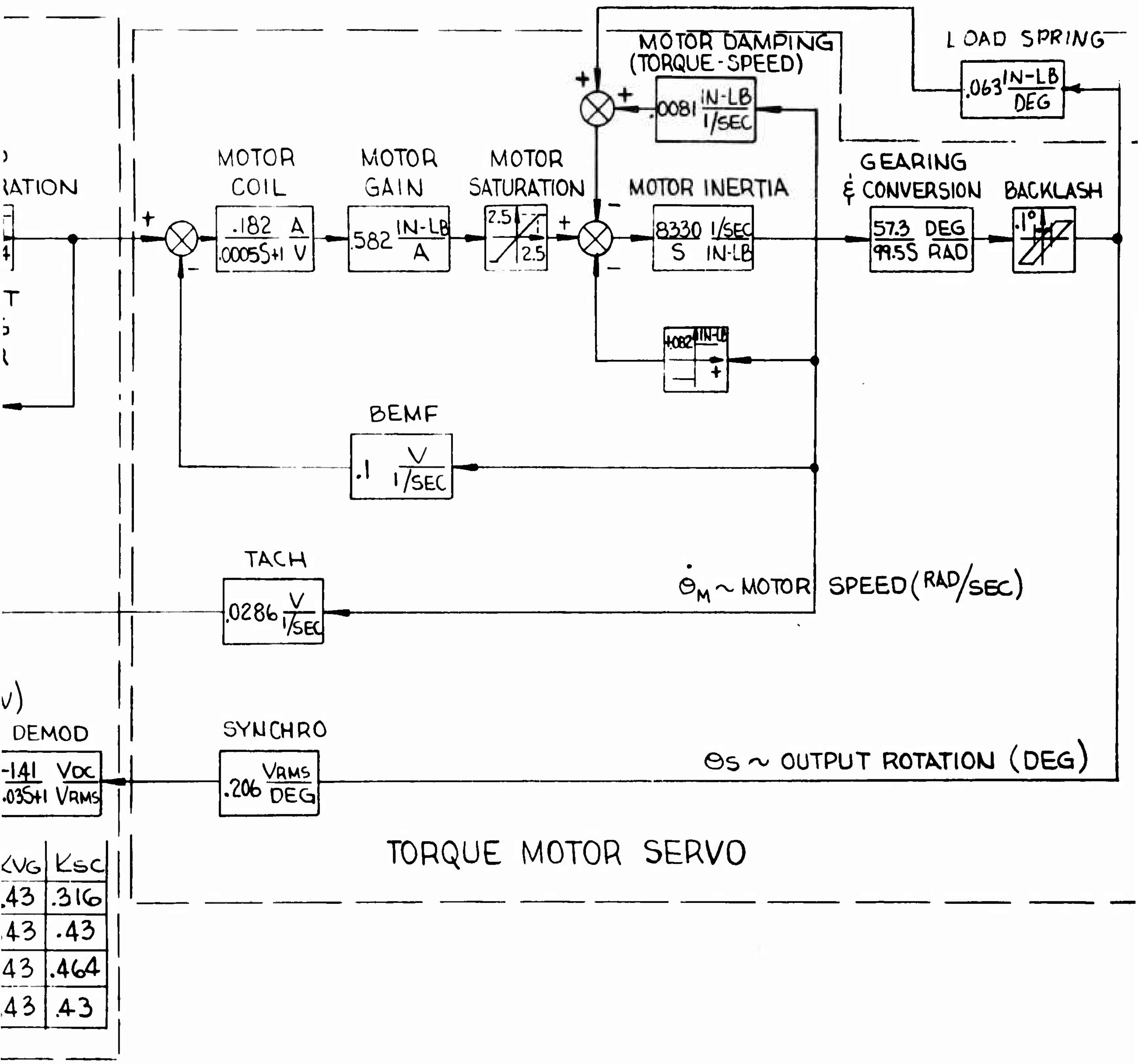
The minor loop computer electronics include the following:



	KFU	KVG	KSC
PITCH	2.2	.43	.316
ROLL	2.2	.43	.43
COLL	2.2	.43	.464
YAW	2.2	.43	.43

Figure 11. Minor Loop Block Diagram.

A



MOTOR DAMPING (TORQUE-SPEED)
.0081 IN-LB / 1/SEC
LOAD SPRING
.063 IN-LB / DEG
MOTOR COIL
.182 / .0005S+1 A/V
MOTOR GAIN
582 IN-LB / A
MOTOR SATURATION
2.5
2.5
MOTOR INERTIA
8330 / S 1/SEC / IN-LB
GEARING & CONVERSION
57.3 / 99.5S DEG / RAD
BACKLASH
.1°
.082 IN-LB
BEMF
.1 V / 1/SEC
TACH
.0286 V / 1/SEC
$\dot{\theta}_M$ ~ MOTOR SPEED (RAD/SEC)
DEMOD
-141 / .035+1 VDC / VRMS
SYNCHRO
.206 VRMS / DEG
θ_S ~ OUTPUT ROTATION (DEG)
TORQUE MOTOR SERVO
KVG
KSC
.43
.316
.43
.43
.43
.464
.43
.43

TABLE VII. DC TORQUE MOTOR SERVO CHARACTERISTICS

Weight	2.94 lb
Volume	3 in. x 3 in. x 4-1/2 in.
Gear Ratio (Motor/Output)	99.57:1
Torque Gain	.58 in.-lb(at motor)/Amp
Stall Torque (Output)	200 in.-lb
No Load Speed	3 rad/sec at output shaft
Backlash (at Output)	.1 deg total (maximum)
Position Feedback	Synchro
Rate Feedback	Tachometer
Synchro Excitation	115V, 400 cps (nominal)
Synchro Output	.206V_{RMS}/deg
Tachometer Output	3V/1000 RPM of motor
Power Input (at Maximum Torque)	100 watts (approximately)
Seal	"O" ring
Clutch	Disconnect type
Connector	Pygmy type
Coil Impedance	5.5 ± 0.7 ohms
Motor Time Constant (L/R)	0.0005 sec
Motor Inertia	.00012 in.-lb/sec^2 at motor
Breakaway Torque	.082 in.-lb(maximum)at motor
Induced Back EMF	Approximately .1VDC/rad/sec of motor

1. A high-gain (and fast) forward path power amplifier (to provide power to drive the motor coil) and a front-end summing amplifier.

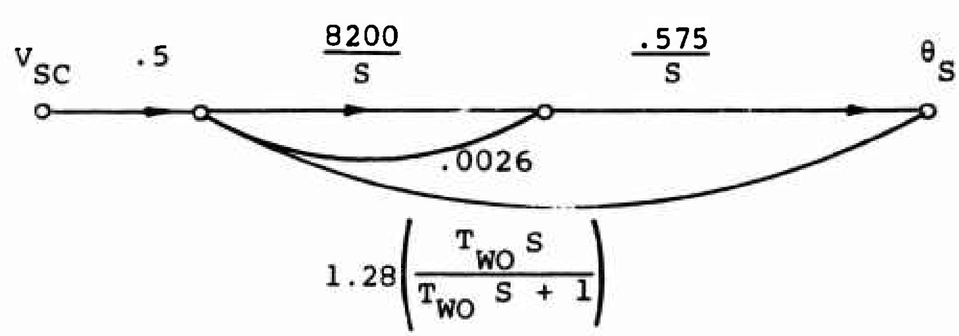
2. A current limiter feedback path which provides stiff limiting of the motor current (by the high gain when the dead zone is exceeded).

3. A tachometer feedback path to provide damping of the position loop.

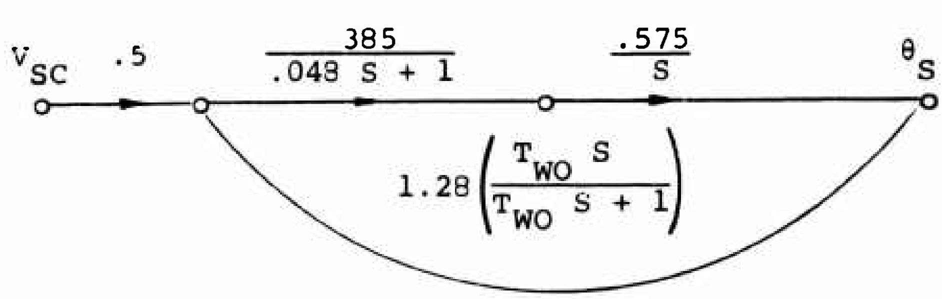
4. A washed-out follow-up path to provide short term positioning and long-term integration.

<u>Simplified Minor Loop Representation</u>

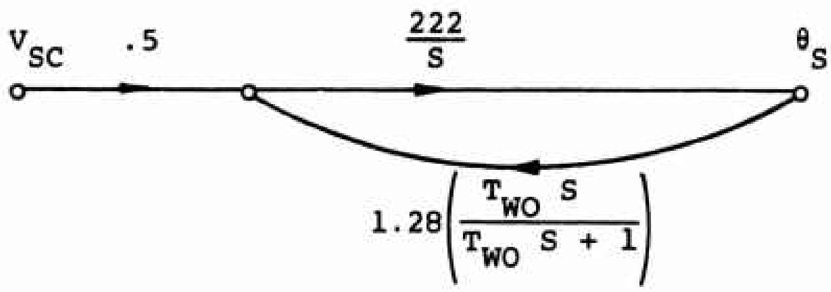
A simplified linear representation of the minor loop, which is useful when analyzing the slower major loop closures, is simply a straight gain and an integration (whose gain is inversely proportional to the washout time constant). This relationship is shown in the following successive signal flow diagram reductions. Neglecting the servo and computer nonlinearities, BEMF, motor damping, load spring and lumping gains, we have (approximately)



We see that the previous signal flow diagram reduces to



Neglecting the short time constant in the above signal flow diagram, we have



We can see that the resulting transfer function for frequencies above $1/T_{WO}$ is approximately

$$\frac{\theta_S}{V_{SC}} \approx \frac{.5}{1.28\left(\frac{T_{WO} S}{T_{WO} S + 1}\right)} = .39\left(1 + \frac{1}{T_{WO} S}\right)$$

Sensor Characteristics

The sensors (outside of those defined as part of the PAS) needed for system operation are as follows:

1. Body axis rate gyros (roll, pitch and yaw)
2. Vertical gyro (roll and pitch attitude)
3. Horizontal situation indicator (heading error)
4. Directional gyro and synchronizer
5. Body axis velocity sensors (longitudinal, lateral and vertical)
6. Attitude stabilized accelerometers (longitudinal and vertical)
7. Body mounted accelerometer (lateral)
8. Radar altimeter
9. Barometric altimeter
10. Barometric rate
11. Rotor or engine rpm sensor

The work performed during this study assumed that the sensor math models could be approximated by straight gains. This first-cut assumption should be modified, if required, by test data at a later date.

From flight tests on a UH-1B run at Fort Rucker and Fort Benning in March 1964, the following ranges of aircraft parameters were obtained for typical Army maneuvers (for an armament helicopter):

1. Rotor rpm - 290 to 339
2. Airspeed - hover to 100 knots
3. Collective stick position - full down to 78 percent
4. Fore and aft cyclic stick - 17 to 80 percent from full aft
5. Lateral cyclic stick - 12 to 90 percent from full left

6. Rudder pedal position - 1.5 to 94 percent from full left

7. Vertical acceleration at C.G. - .5 to 1.88 g

8. Lateral acceleration at C.G. - ±.35 g

9. Fore and aft acceleration at C.G. - ±.13 g

10. Roll rate - ±49 degrees per second

11. Pitch rate - ±18.5 degrees per second

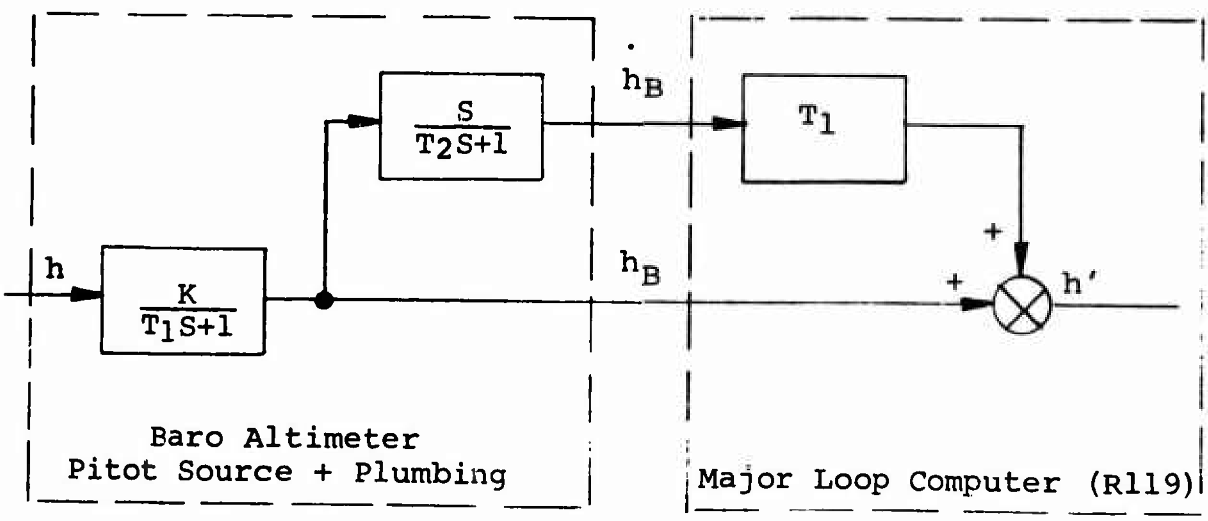
12. Yaw rate - ±47 degrees per second

13. Pitch attitude - 23 degrees nose up, 30 degrees nose down

14. Roll attitude - ±62 degrees

The above parameter ranges define minimum sensor ranges.

Complementary filtering of sensors can be used if test data on the sensors indicate potential problems due to inherent sensor lags, vibration sensitivity, attitude or angle of attack sensitivity, etc. The following block diagram indicates a filtering technique that can be used:



$T_1 >> T_2$

From the above diagram, we have

$$h_B = \frac{K}{T_1 S+1}\, h$$

$$\dot{h}_B = \frac{K}{T_1 S+1}\,\frac{S}{T_2 S+1}\, h \approx \frac{K\ S}{T_1\ S+1}$$

Therefore, we have

$$h' = T_1 \dot{h}_B + h_B$$

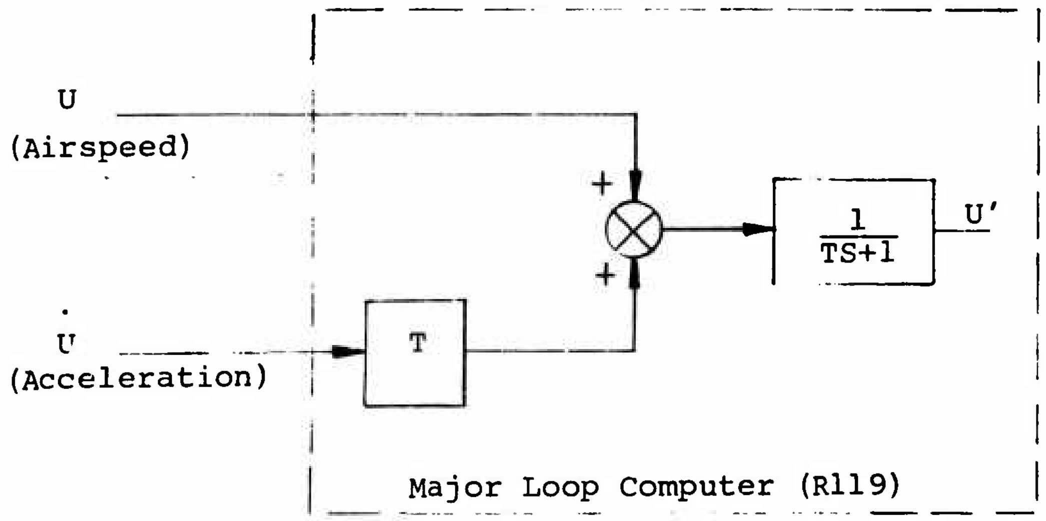
$$h' \approx \left(\frac{KT_1 S}{T_1 S+1}\right) + \frac{K}{T_1 S+1} \quad h$$

$$h' \approx K\,\frac{T1ST1}{T_1 ST1}\, h$$

$$h' \approx Kh$$

An $\dot{h}_B$ path has been provided in the major loop computer to implement the complementary filtering, if required.

The technique of complementing adjacent derivative sensors can be used to combine raw accelerometer and airspeed information to obtain a modified airspeed signal. The following block diagram illustrates the technique that can be implemented (not shown on the axis block diagrams).



From the above diagram, we have

$$u' = \left(\frac{TS+1}{TS+1}\right) u = u$$

In the above mechanization, the low frequency information is being supplied by the airspeed sensor and the high-frequency information is being supplied by the accelerometer. Using an accelerometer that is not altitude stabilized would require inserting a long time constant washout in the accelerometer leg to remove the steady-state output (due to orientation) of the accelerometer.

ANALYSIS DESIGN TOOLS AND RESULTS

The analysis design tools that were used in this study consisted of the following:

1. IBM digital simulation program called Continuous System Modeling Program (CSMP).

2. ANC root locus program.

3. ANC breadboard setup consisting of:

    a. PAS computer.

    b. Hardware servo, load stand and servo electronics.

    c. Analog simulation of the aircraft and sensors.

There are several CSMP programs that have been generated for the following design applications:

1. A simplified longitudinal 3-degree-of-freedom (DOF) simulation (PAS and aircraft representation) to:

    a. Check basic uncoupled aircraft time response.

    b. Observe system response in each of the longitudinal modes.

c. Examine the effects of external disturbances on the system.

2. A simplified lateral 3-DOF simulation similar to 1.

3. A simplified lateral 3-DOF simulation to study turn coordination performance.

4. A complex (PAS and longitudinal) 3-DOF simulation to:

   a. Examine the response of the system to pilot force inputs.

   b. Check the effects of mode switching on the system.

   c. Use for system testing.

5. A complex lateral 3-DOF simulation similar to 4.

6. A simplified PAS, 6-DOF simulation to:

   a. Check the cross coupling effects of the basic aircraft.

   b. Check the effects of using decoupling feedbacks in the PAS.

7. A detailed simulation of the servo loop (including load effects.

Digital Simulation Programs

Simplified Longitudinal CSMP Description

A block diagram of a simplified longitudinal CSMP simulation is shown in Figure 12. A description of the airframe 3-DOF equations of motion (with parameter designations written like the computer program) is shown on Table VIII. Tables IX through XI tabulate the aircraft stability derivatives for flight conditions 1, 4 and 12, respectively.

The CSMP block diagram (Figure 12) has been set up with sufficient generality such that parameter changes can be made without having to recompile the program to make a new run. System inputs can be either: (1) low passed gaussian noise signal that is inserted as an airspeed or angle of attack gust, or (2) a pulse input (with variable amplitude, duration and starting point) that is inserted as a force type input (through CTST2 of CTST4) or into the free aircraft (through CTST1 or CTST3).

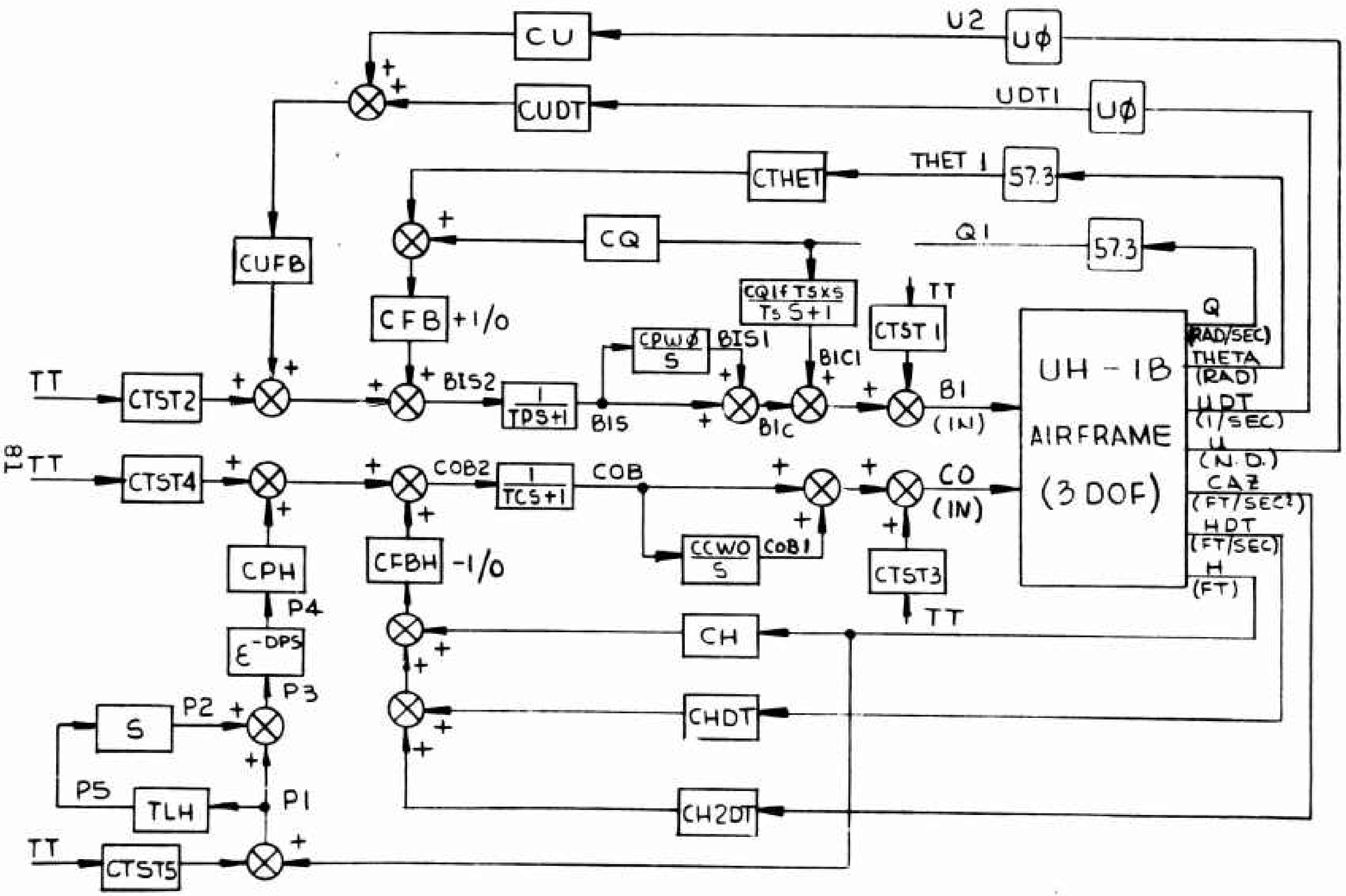


Figure 12. Simplified Longitudinal CSMP Block Diagram.

TABLE VIII. LONGITUDINAL AIRCRAFT EQUATIONS (3-DOF)

```
UDT=XU*U+XW*ALFA+UXQ*Q-GUO*THETA+UXB1*B1+UXCO*CO
+XU*UG+XW*ALFAG

ALFADT=ZU*U+XW*ALFA+UZQ*Q-GUOS*THETA+UZB1*B1+UZCO*CO
+ZU*UG+ZW*ALFAG

THE2DT=AMU*U+AMW*ALFA+UMQ*Q+UMB1*B1+UMCO*CO
+AMU*UG+AMW*ALFAG

U=INTGRL(0.0,UDT)

ALFA=INTGRL(0.0,ALFADT)

THEDT=INTGRL(0.0,KIY*THE2DT)

THETA=INTGRL(0.0,THEDT)

Q=THEDT
```

Sensor Equations

```
Q1=57.3*Q

THET1=57.3*THETA

U2=UO*U

UDT1=UO*UDT

HDT=U2*(THETA-ALFA)

CAZ=U2*(Q-ALFADT)+ALX*THE2DT

H=INTGRL(0.0,HDT)
```

TABLE IX. 3-DOF LONGITUDINAL STABILITY DERIVATIVES (FC1)

Flight Condition 1

U_o = 80 kt = 135.8 ft/sec

W = 6750 lb ; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = -15.2 ft/sec

θ_o = -6.47 deg; $\sin \frac{\theta_o}{57.3}$ = -.113

Alt = 3000 ft

C.G. = 134.4 in.

XU = X_u/m = .048	ZU = Z_u/m = -.0867	AMU = $U_o M_u/I_y$ = -.377
XW = X_w/m = -.081	ZW = Z_w/m = -1.125	AMW = $U_o M_w$ = +1.965
UXQ = $X_q/mU_o - \frac{W_o}{U_o}$ = +.122	UZQ = $\left(1 + \frac{Z_q}{mU_o}\right)$ = +.986	UMQ = M_q/I_y = -.513
GUO = G/U_o = +.237	GUOS = $\frac{G}{U_o}\sin(\theta_o)$ = -.027	
UXB1 = X_{B1}/mU_o = +.012	UZB1 = Z_{B1}/mU_o = +.040	UMB1 = M_{B1}/I_y = -.313
UXCO = X_{CO}/mU_o = -.011	UZCO = Z_{CO}/mU_o = -.161	UMCO = M_{CO}/I_y = +.320

TABLE X. 3-DOF LONGITUDINAL STABILITY DERIVATIVES (FC4)

Flight Condition 4

U_o = 40 kt = 67.7 ft/sec

W = 6750 lb ; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = -2.23 ft/sec

θ_o = -1.881 deg; $\sin \frac{\theta_o}{57.3} = -.033$

Alt = 3000 ft

C.G. = 134.4 in.

XU = X_u/m = -.029	ZU = Z_u/m = -.163	AMU = $U_o M_u/I_y$ = -.053
XW = X_w/m = -.018	ZW = Z_w/m = -.900	AMW = $U_o M_w$ = +.183
UXQ = $X_q/mU_o - \frac{W_o}{U_o}$ = +.0228	UZQ = $\left(1 + \frac{Z_q}{mU_o}\right)$ = +.999	UMQ = M_q/I_y = -.385
GUO = G/U_o = +.474	GUOS = $\frac{G}{U_o}\sin(\theta_o)$ = -.023	
UXB1 = X_{B1}/mU_o = +.019	UZB1 = Z_{B1}/mU_o = +.033	UMB1 = M_{B1}/I_y = -.215
UXCO = X_{CO}/mU_o = -.0069	UZCO = Z_{CO}/mU_o = -.260	UMCO = M_{CO}/I_y = +.075

TABLE XI. 3-DOF LONGITUDINAL STABILITY DERIVATIVES (FC12)

<u>Flight Condition 12</u>

U_o = 2 kt = 3.40 ft/sec

W = 6750 lb ; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 938 slug-ft^2

W_o = .000132 ft/sec

θ_o = +.219 deg ; $\sin \frac{\theta_o}{57.3}$ = .000066

Alt = 3000 ft

C.G. = 134.4 in.

XU = X_u/m = -.013	ZU = Z_u/m = -.168	AMU = U_oM_u/I_y = +.0057
XW = X_w/m = +.004	ZW = Z_w/m = -.450	AMW = U_oM_w = -.005
UXQ = $X_q/mU_o - \frac{W_o}{U_o}$ = +.172	UZQ = $\left(1 + \frac{Z_q}{mU_o}\right)$ = +1.213	UMQ = M_q/I_y = -.108
GUO = G/U_o = +9.48	GUOS = $\frac{G}{U_o}\sin(\theta_o)$ = +.038	
UXB1 = X_{B1}/mU_o = +.370	UZB1 = Z_{B1}/mU_o = +.0546	UMB1 = M_{B1}/I_y = -.205
UXCO = X_{CO}/mU_o = +.017	UZCO = Z_{CO}/mU_o = -4.71	UMCO = M_{CO}/I_y = -.009

Additional features of the program include:

1. The capability to run various combinations of feedbacks (including the provision to check the free aircraft).

2. The capability to make a succession of runs having increasing system complexity in a single pass on the computer.

3. A model of the pilot for hover height control evaluation.

4. Inclusion of a series servo model to investigate the advantages of a series/parallel servo combination.

Simplified Longitudinal CSMP Results

Comparison of the free aircraft response with USAAVLABS response at F.C. 1 is shown in Figure 13. This figure also shows the breadboard analog response. These responses (to approximately the same pulse input) are fairly close to each other for the first 1.8 seconds. In order to get the ANC traces shown in Figure 13, the pitch inertia, Iy, had to be increased by 25 percent over that value indicated by the C-81 frequency response data. This indicates that perhaps there is a difference between the C-81 frequency response data (which was used to get the ANC model) and C-81 time response results. The dispersion after 1.8 seconds is assumed to be due to:

1. Large attitudes that are not compensated for in the ANC linearization.

2. Additional effects (like rotor system) in the C-81 time response program that are not reflected in the C-81 stability derivative output.

It was assumed that the model used to obtain the match shown in Figure 13 was adequate for subsequent flight control design.

The Appendix contains the digital program listing for the system described in Figure 12 and Tables VIII through XI.

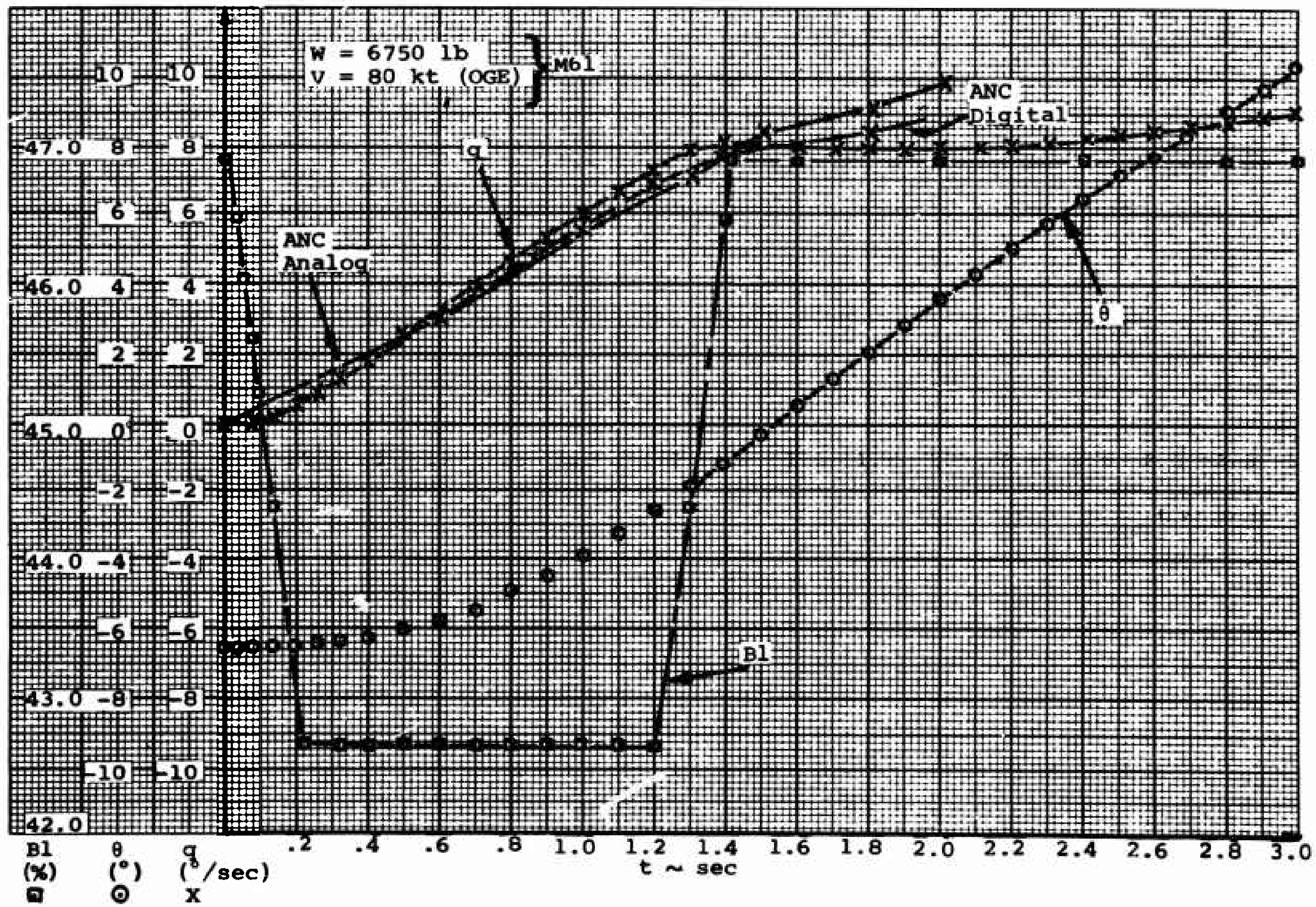


Figure 13. Free Aircraft Response to .5-Inch Aft B1 Pulse (FC1).

Simplified Lateral CSMP Description

A block diagram of the simplified lateral CSMP simulation is shown in Figure 14. A description of the airframe 3-DOF equations of motion (with parameter designations written like computer programs) is shown in Table XII. Tables XIII through XV tabulate the aircraft stability derivatives for flight conditions 1, 4 and 12, respectively.

The same sort of program flexibility has been incorporated in the simplified lateral CSMP as was used in the longitudinal CSMP.

Simplified Lateral CSMP Results

Figures 15 and 16 show comparisons of the free aircraft pulse responses (obtained from the CSMP simulation of Figure 14) with their USAAVLABS counterparts (C-81 time response). These figures show fairly good correlation using the same roll inertia, I_X, that was used for the C-81 input data.

The appendix contains the digital program listing for the system described by Figure 14 and Tables XII through XV.

Lateral Turn Coordination CSMP Description

Block diagrams of the roll and yaw axes of the lateral turn coordination CSMP simulation are shown in Figures 17 and 18, respectively. Descriptions of the airframe 3-DOF equations of motion are given in Table XII. Tables XIII through XV tabulate the aircraft stability derivatives for flight conditions 1, 4, and 12, respectively.

The PAS block diagrams (Figures 17 and 18) are basically complex lateral block diagrams that have been reduced to include only the applicable turn coordination loops.

The yaw axis hardware mechanization has been simplified from that shown in Figure 18. This simplification has not been reflected in Figure 18. Figure 18 was included to be used as a base from which future modifications can be made.

Complex Longitudinal CSMP Description

Block diagrams of the pitch and collective axes of the complex longitudinal CSMP simulation are shown in Figures 19 and 20, respectively. Descriptions of the airframe 3-DOF equations of motion are shown in Table VIII. Tables IX through XI tabulate the aircraft stability derivatives for flight conditions 1, 4 and 12, respectively.

The PAS axis block diagrams (Figures 19 and 20) have been set up to obtain a near one-to-one correlation with the PAS. This has been done to investigate the effects of

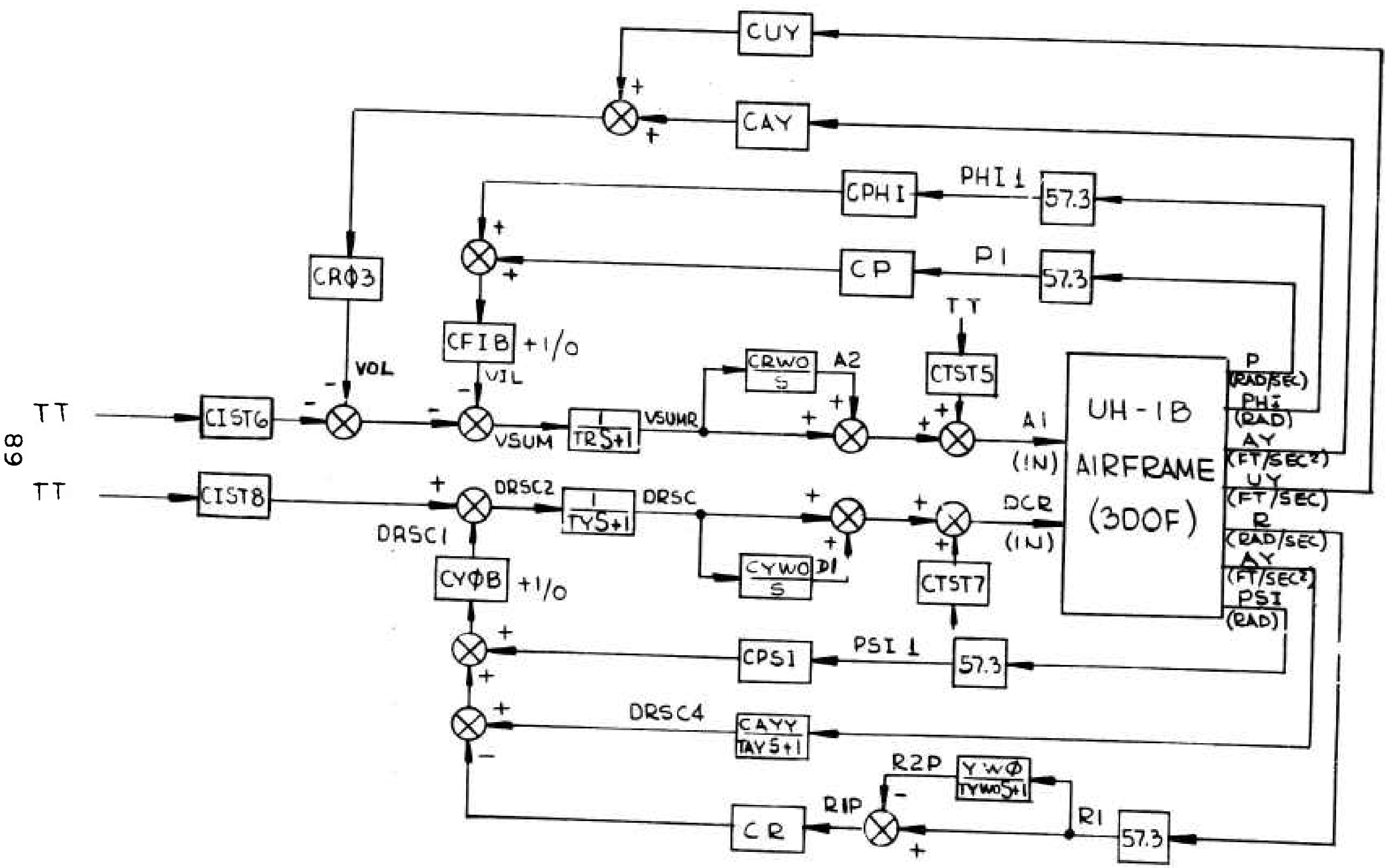


Figure 14. Simplified Lateral CSMP Block Diagram.

TABLE XII. LATERAL AIRCRAFT EQUATIONS (3-DOF)

```
BETADT=YU*BETA+UYP*P+GUO*PHI+UYR*R+UYA1*A1+UYDLR*DLR
+YV*BETAG

PHI2DP=ALV*BETA+ULP*P+UIXZ*RDTP+ULR*R+ULA1*A1+ULDLR*
DLR+ALV*BETAG+ULP*PG

RDT=ANV*BETA+UNP*P+UIZZ*PHI2DT+ANR+R+UNA1*A1+UNDLR*
DLR+ANV*BETAG+UNP*PG

PHI2DT=KIX*PHI2DP

RDTP=REALPL(0.0,TRDT,RDT)

BETA=INTGRL(0.0,BETADT)

PHIDT=INTGRL(0.0,PHI2DT)

PHI=INTGRL(0.0,PHIDT)

P=PHIDT

R=INTGRL(0.0,RDT)

PSI=INTGRL(0.0,R)
```

<u>Sensor Equations</u>

```
R1=57.3*R

PHI1=57.3*PHI

AY=UO*(YV*BETA+UYDLR*DLR)+ALY*RDTP

UY=UO*BETA

PSI1=57.3*PSI
```

TABLE XIII. 3-DOF LATERAL STABILITY DERIVATIVES (FC1)

Flight Condition 1

U_o = 80 kt = 135.8 ft/sec

W = 6750 lb ; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = -15.2 ft/sec

θ_o = -6.57 deg ; sin $\frac{\theta_o}{57.3}$ = -.113

Alt = 3000 ft

C.G. = 134.4 in.

YV = Y_v/m = -.5211	ALV = $U_o L_v/I_x$ = -14.18	ANV = $U_o N_v/I_z$ = +5.423
UYP = $Y_p/mU_o + \frac{W_o}{U_o}$ = -.127	ULP = L_p/I_x = -3.980	UNP = N_p/I_z = -.0819
GUO = G/U_o = +.237	UIXZ = I_{xz}/I_x = +1.411	UIZZ = I_{xz}/I_z = -.1317
UYR = $\left(-1 + \frac{Y_r}{mU_o}\right)$ = -.985	ULR = L_r/I_x = +2.856	ANR = N_r/I_z = -1.388
UYA1 = Y_{A1}/mU_o = +.00834	ULA1 = L_{A1}/I_x = +1.826	UNA1 = N_{A1}/I_z = -.0012
UYDLR = Y_{DLR}/mU_o = +.0135	ULDLR = L_{DLR}/I_x = +3.117	UNDLR = N_{DLR}/I_z = -1.373

TABLE XIV. 3-DOF LATERAL STABILITY DERIVATIVES (FC4)

Flight Condition 4

U_o = 40 kt = 67.7 ft/sec

W = 6750 lb ; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = -223 ft/sec

θ_o = 1.881 deg ; $\sin \frac{\theta_o}{57.3}$ = -.033

Alt = 3000 ft

C.G. = 134.4 in.

YV = Y_v/m = -.274	ALV = U_oL_v/I_x = -.535	ANV = U_oN_v/I_x = +1.839
UYP = $Y_p/mU_o + \frac{W_o}{U_o}$ = -.027	ULP = L_p/I_x = -3.703	UNP = N_p/I_z = -.261
GUO = G/U_o = +.474	UIXZ = I_{xz}/I_x = +1.41	UIZZ = I_{xz}/I_z = +.132
UYR = $\left(-1 + \frac{Y_r}{mU_o}\right)$ = -.980	ULR = L_r/I_x = +1.993	ANR = N_r/I_z = -.949
UYA1 = Y_{A1}/mU_o = +0.123	ULA1 = L_{A1}/I_x = +1.760	UNA1 = N_{A1}/I_z = -.00053
UYDLR = Y_{DLR}/mU_o = +.019	ULDLR = L_{DLR}/I_x = -2.20	UNDLR = N_{DLR}/I_z = -.9660

TABLE XV. 3-DOF LATERAL STABILITY DERIVATIVES (FC12)

Flight Condition 12

U_o = 2 kt = 3.40 ft/sec

W = 6750 lb ; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 938 slug-ft^2

W_o = .00032 ft/sec

θ_o = +.219 deg ; $\sin \frac{\theta_o}{57.3}$ = .000066

Alt = 3000 ft

C.G. = 134.4 in.

YV = Y_v/m = -.05	ALV = $U_o L_v/I_x$ = -.219	ANV = $U_o N_v/I_z$ = +.044
UYP = $Y_p/mU_o + \frac{W_o}{U_o}$ = -.207	ULP = L_p/I_x = -1.44	UNP = N_p/I_z = +.059
GUO = G/U_o = +9.43	UIXZ = I_{xz}/I_x = 1.41	UIZZ = I_{xz}/I_z = +.132
UYR = $\left(-1 + \frac{Y_r}{mU_o}\right)$ = -.826	ULR = L_r/I_x = +.996	ANR = N_r/I_z = -.420
UYA1 = Y_{A1}/mU_o = +.244	ULA1 = L_{A1}/I_x = +1.76	UNA1 = N_{A1}/I_z = -.0023
UYDLR = Y_{DLR}/mU_o = +.384	ULDLR = L_{DLR}/I_x = +2.22	UNDLR = N_{DLR}/I_z = -.971

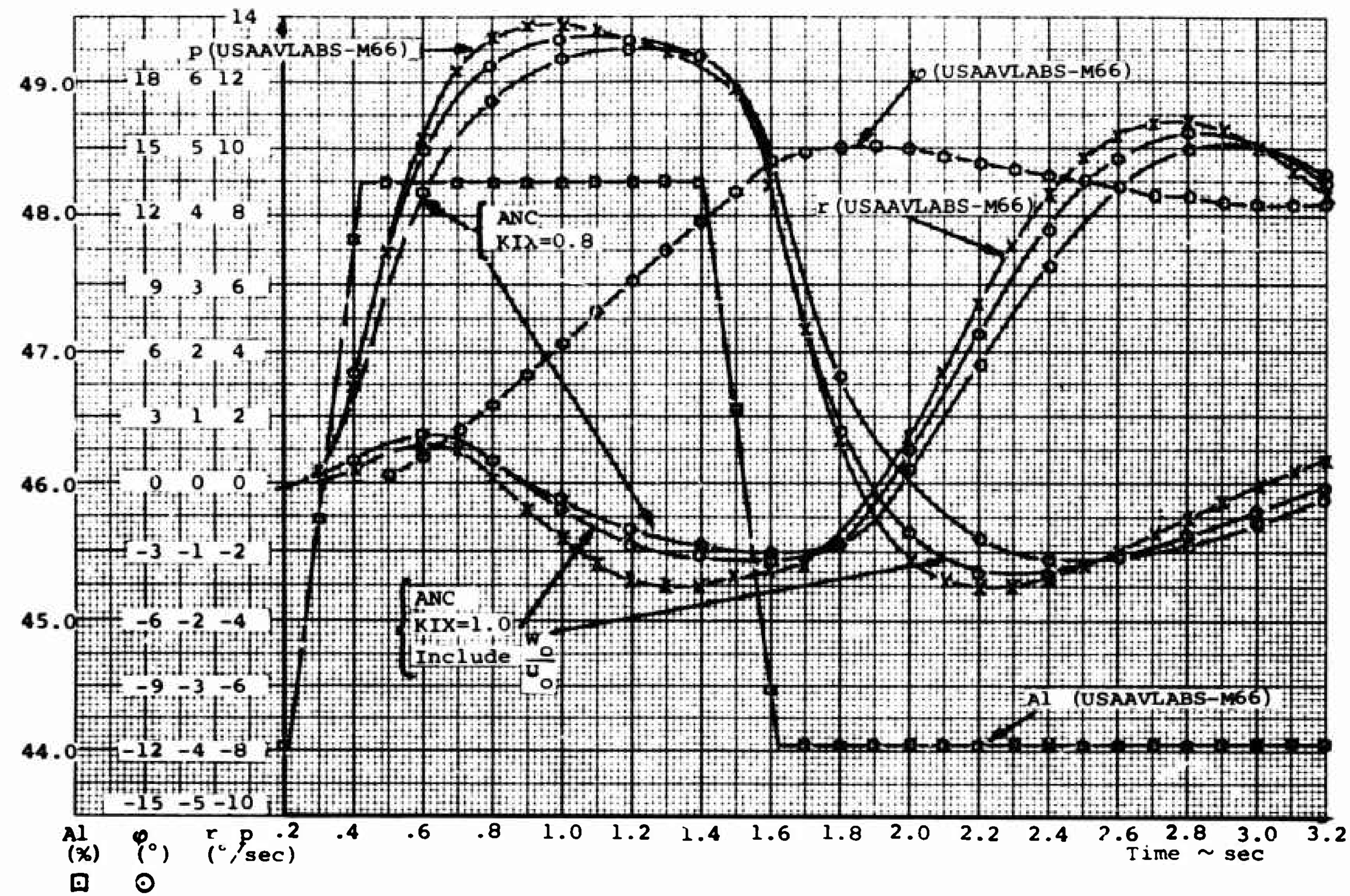


Figure 15. Free Aircraft Response to .5-Inch Right Lat. Cyclic Pulse (FC1).

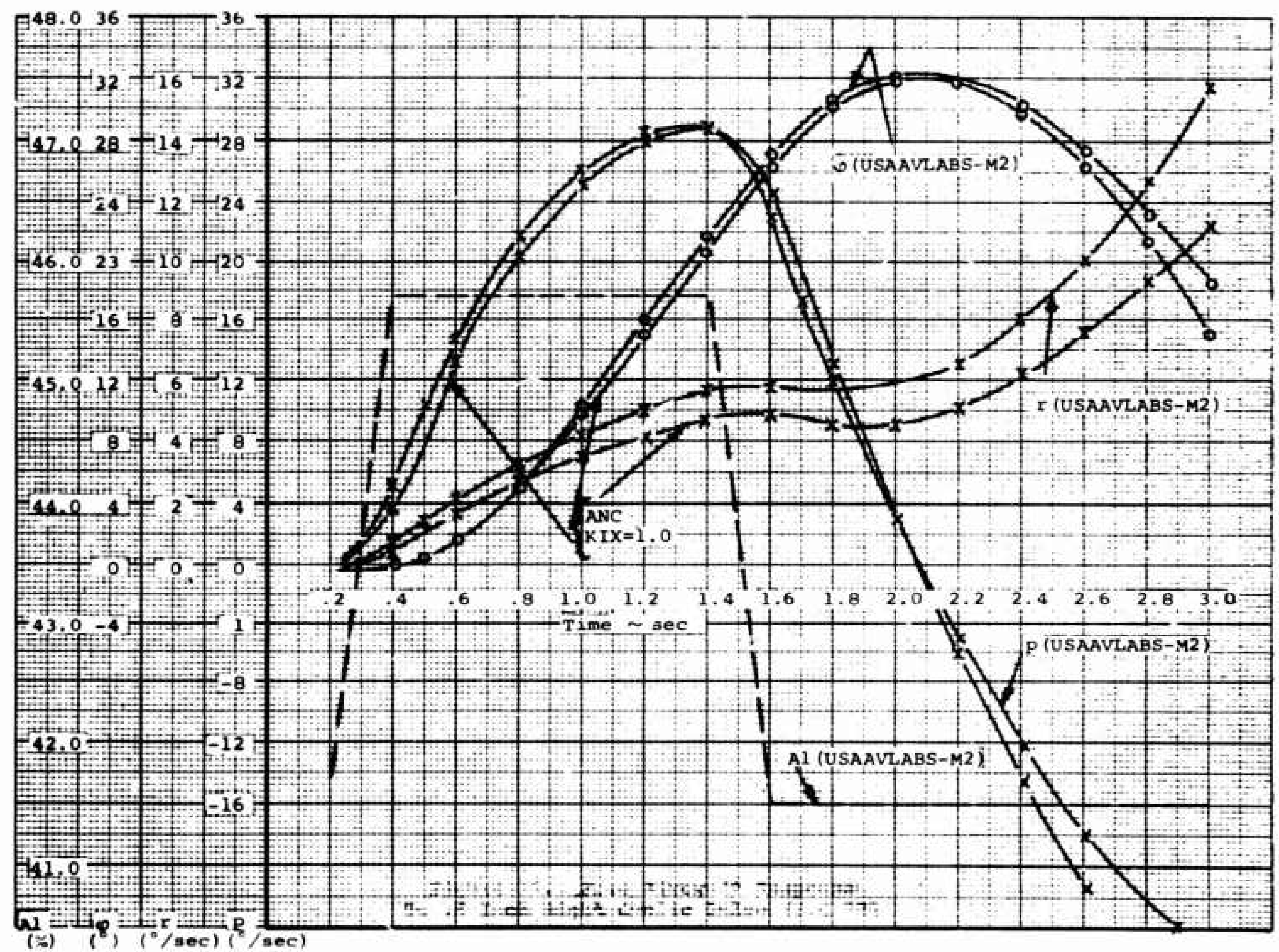


Figure 16. Free Aircraft Response to .5-Inch Right Cyclic Pulse (FC12).

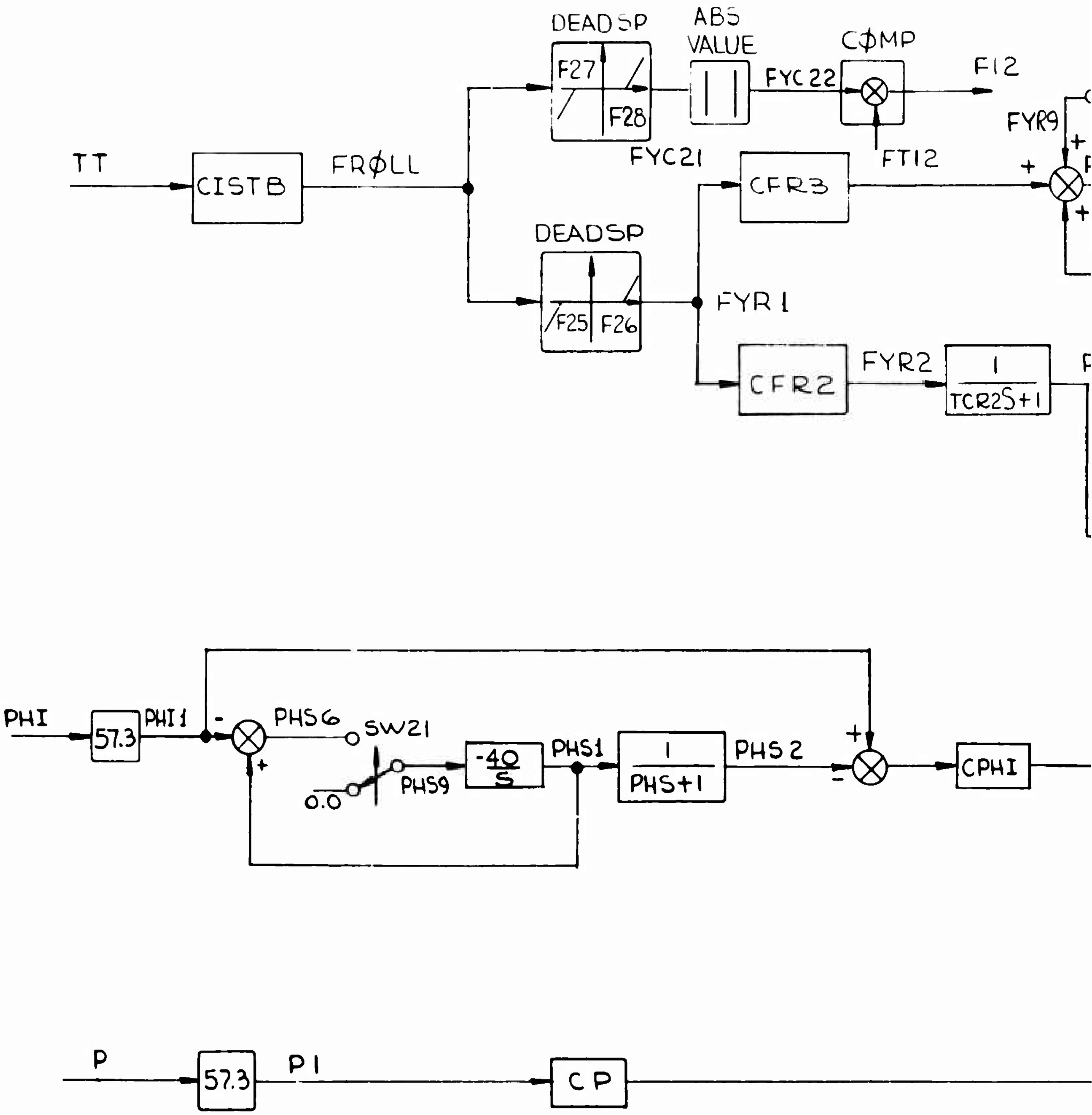
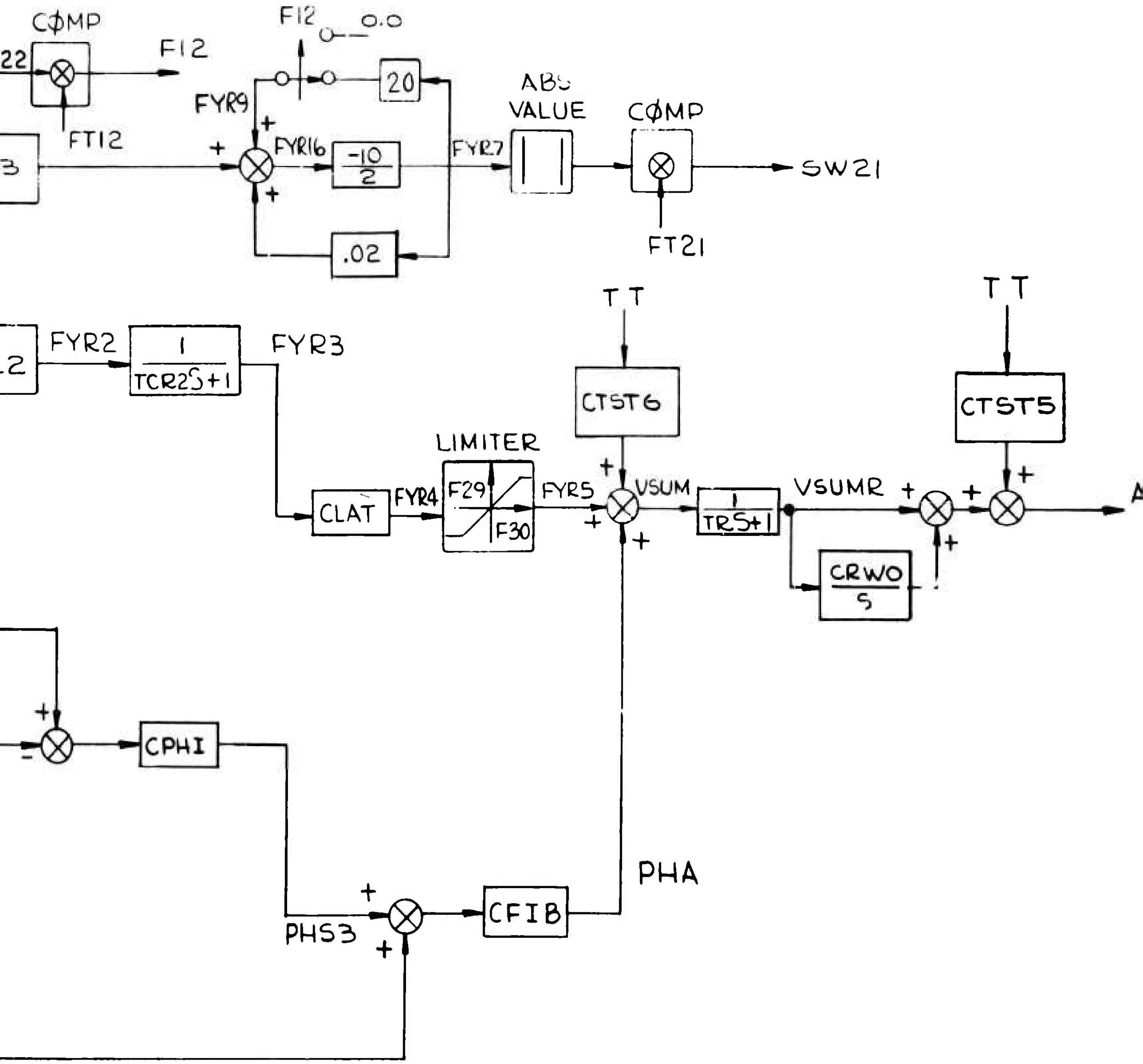


Figure 17. Roll Axis Turn Coordination CSMP Diagram.



CØMP
22
F12
FT12
F12
0.0
20
ABS
VALUE
CØMP
FYR9
+
FYR16
-10
2
FYR7
SW21
FT21
3
+
+
.02
T T
T T
FYR2
1
TCR2S+1
FYR3
CTST6
CTST5
LIMITER
CLAT
FYR4
F29
F30
FYR5
VSUM
1
TRS+1
VSUMR
CRWO
S
CPHI
PHA
PHS3
CFIB

ram.

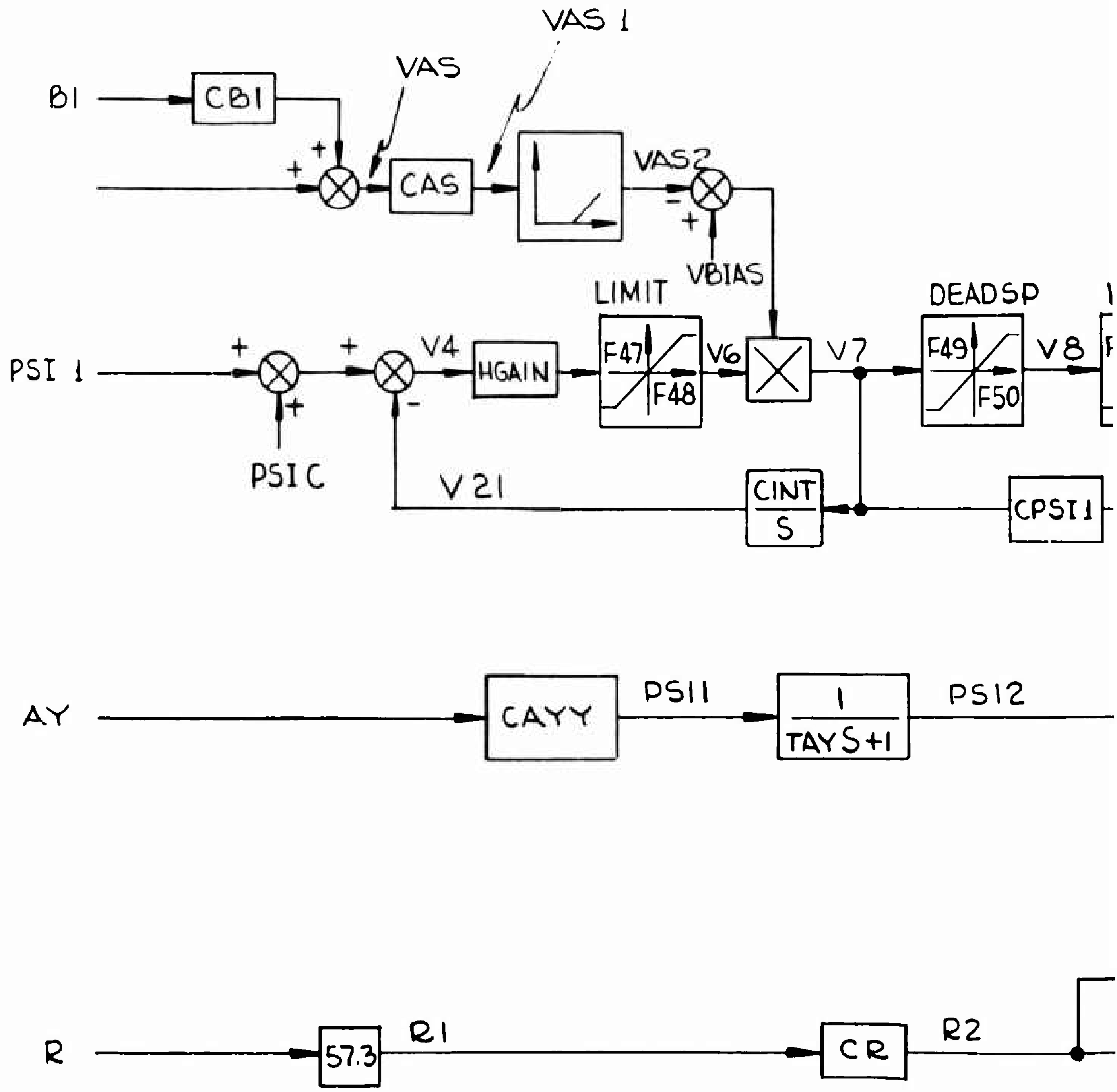
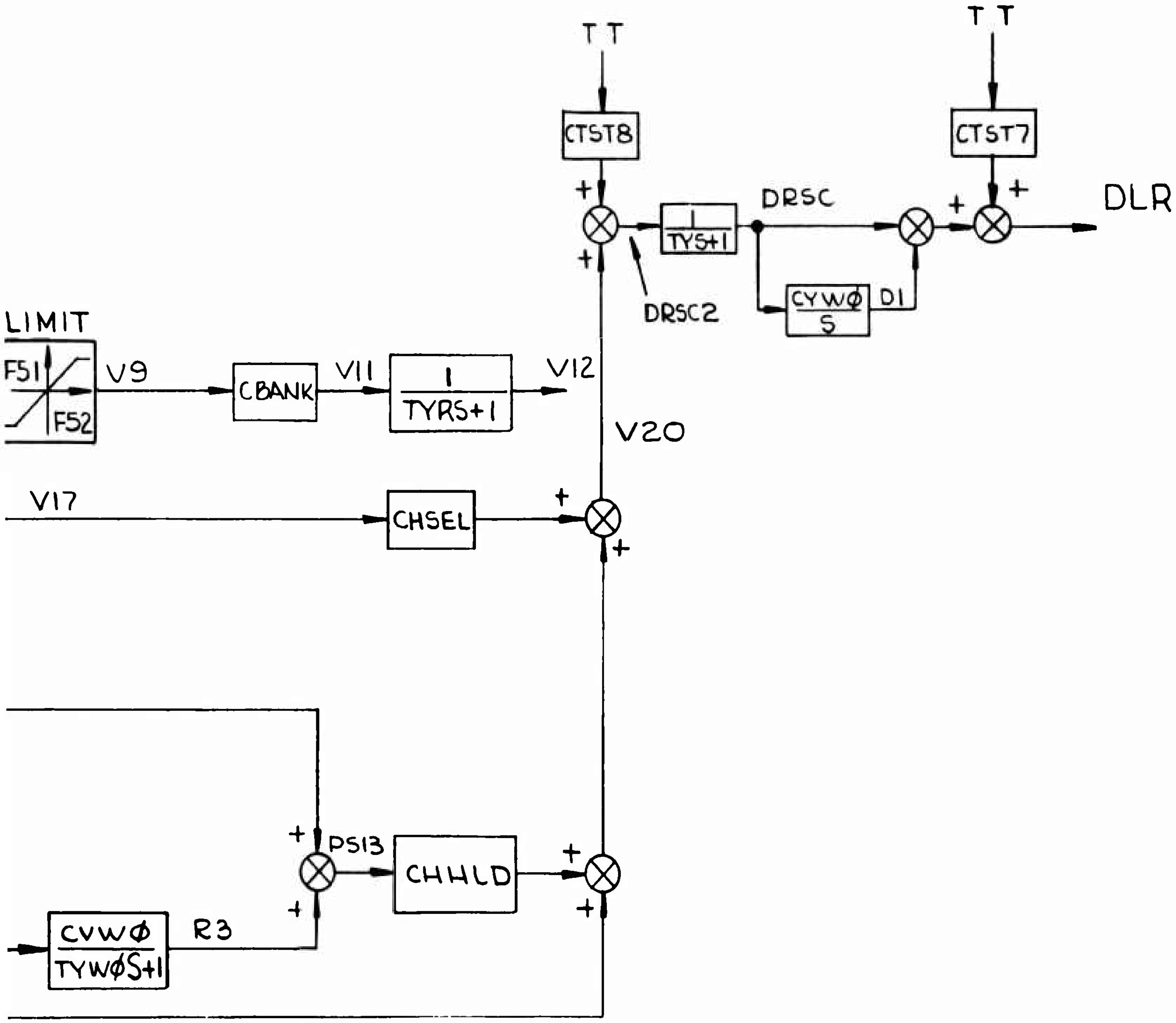


Figure 18. Yaw Axis Turn Coordination CSMP Diagram.


T T
CTST8
DRSC
1
TYS+1
DRSC2
CYWØ
S
D1
T T
CTST7
DLR
LIMIT
F51
F52
V9
CBANK
V11
1
TYRS+1
V12
V20
V17
CHSEL
PSI3
CHHLD
CVWØ
TYWØS+1
R3

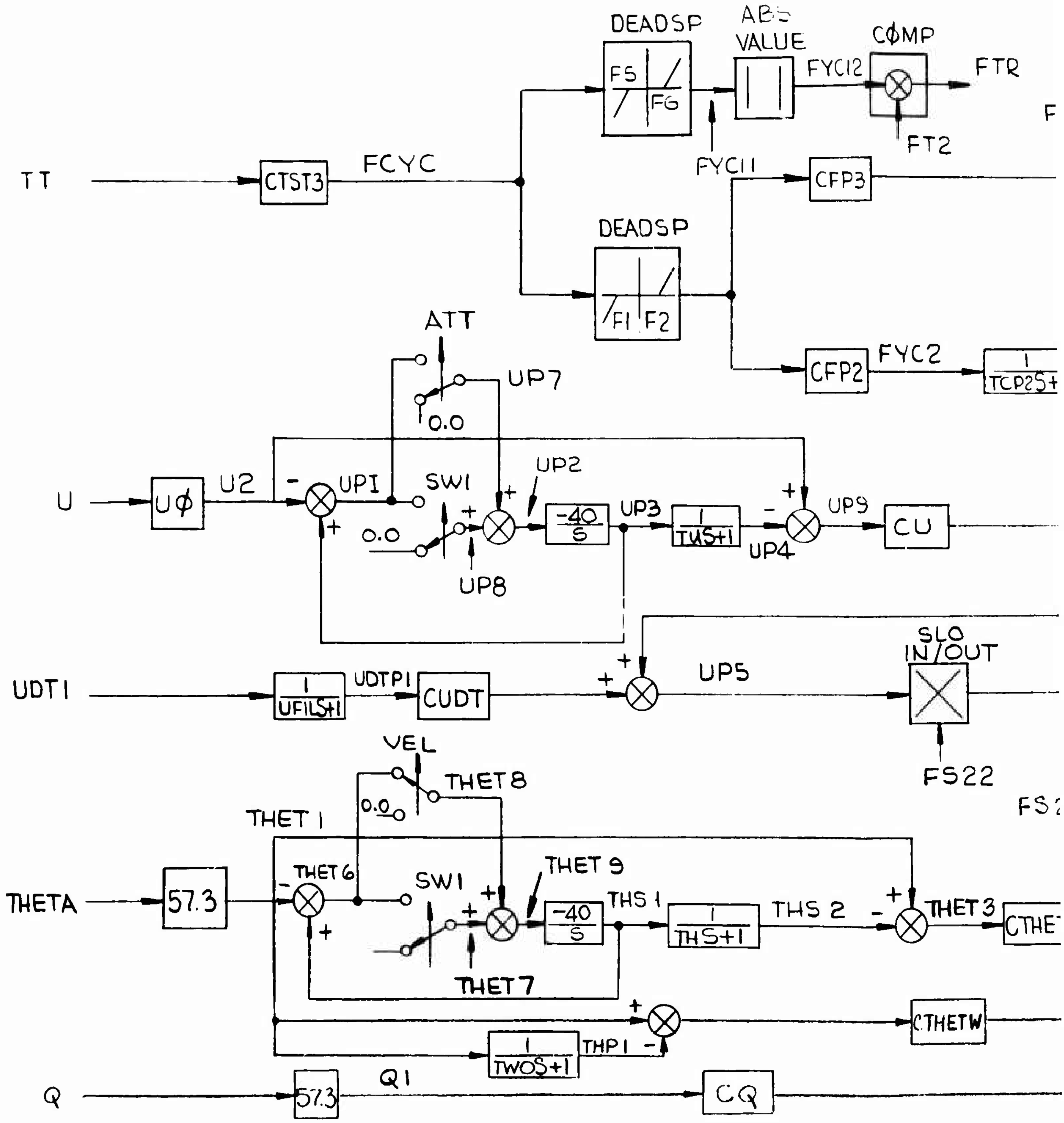
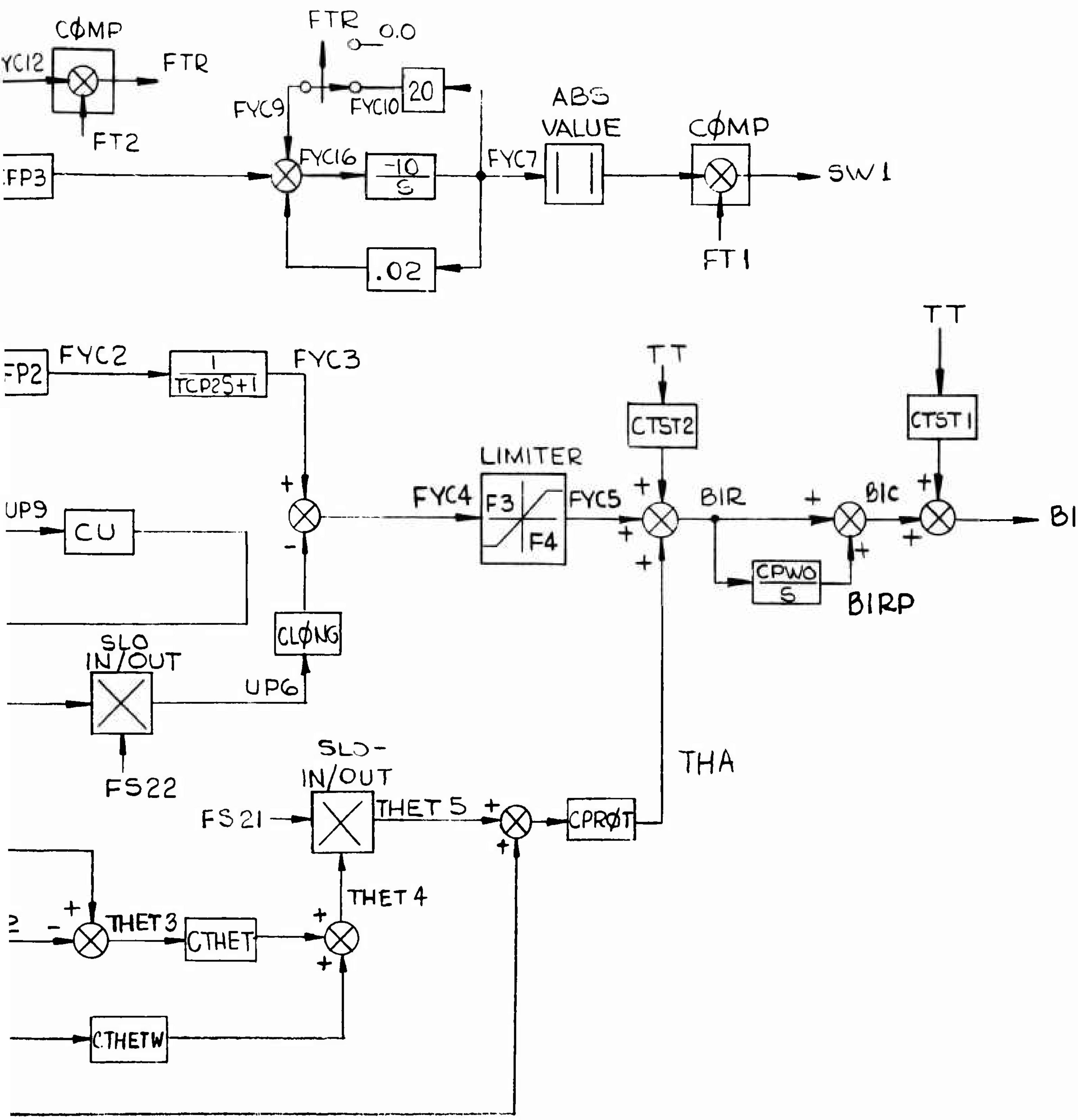


Figure 19. Pitch Axis CSMP Diagram.

A



CØMP
YC12
FTR
FT2
FTR
0.0
20
FYC9
FYC10
FYC16
-10
S
.02
FP3
FYC7
ABS
VALUE
CØMP
SW1
FT1
FP2
FYC2
1
TCP2S+1
FYC3
UP9
CU
LIMITER
FYC4
F3
F4
FYC5
TT
CTST2
BIR
TT
CTST1
BIc
BI
CPWO
S
BIRP
CLØNG
SLO
IN/OUT
UP6
FS22
THA
SLO-
IN/OUT
FS21
THET 5
CPRØT
THET 4
THET3
CTHET
CTHETW

3

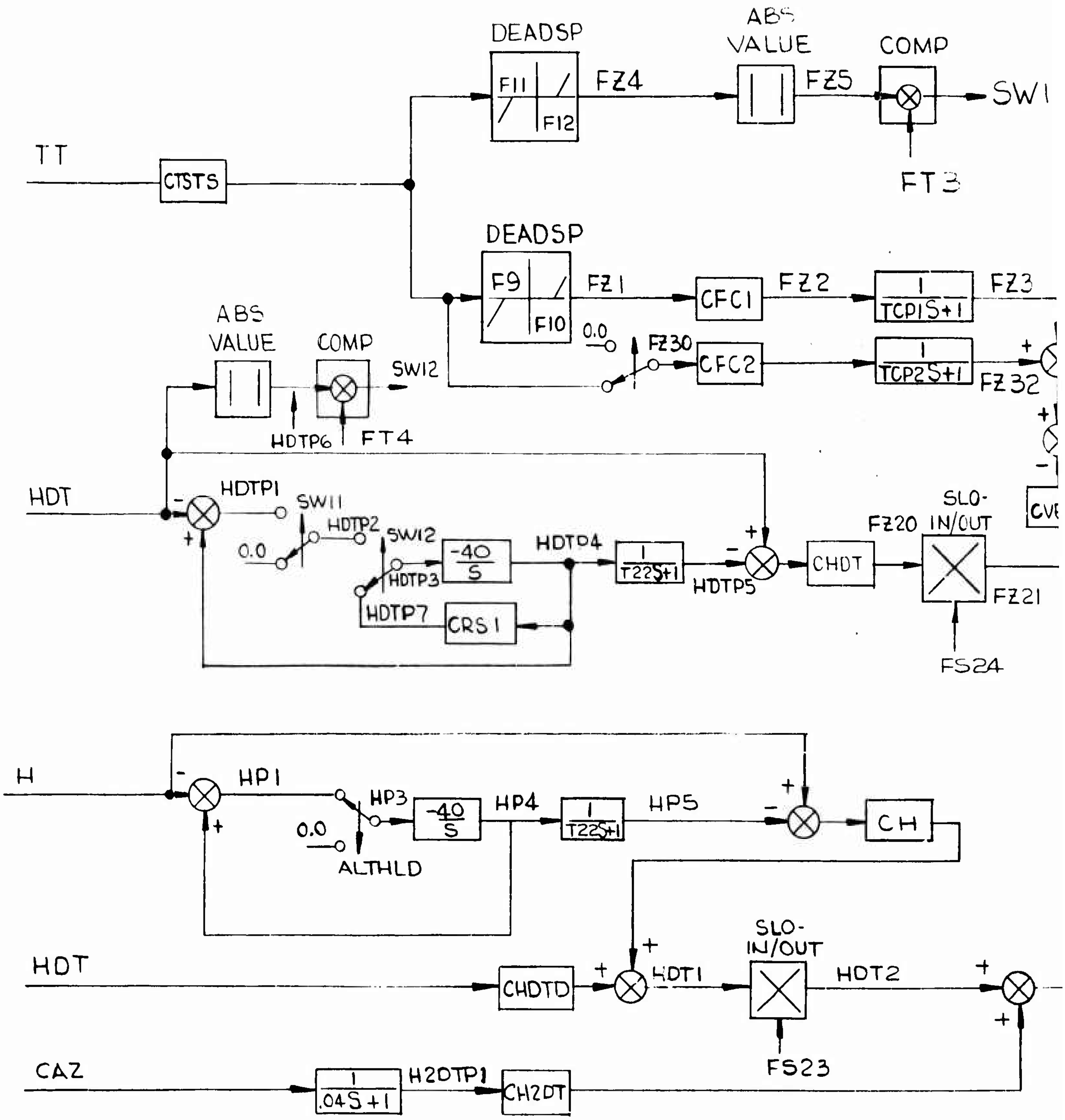
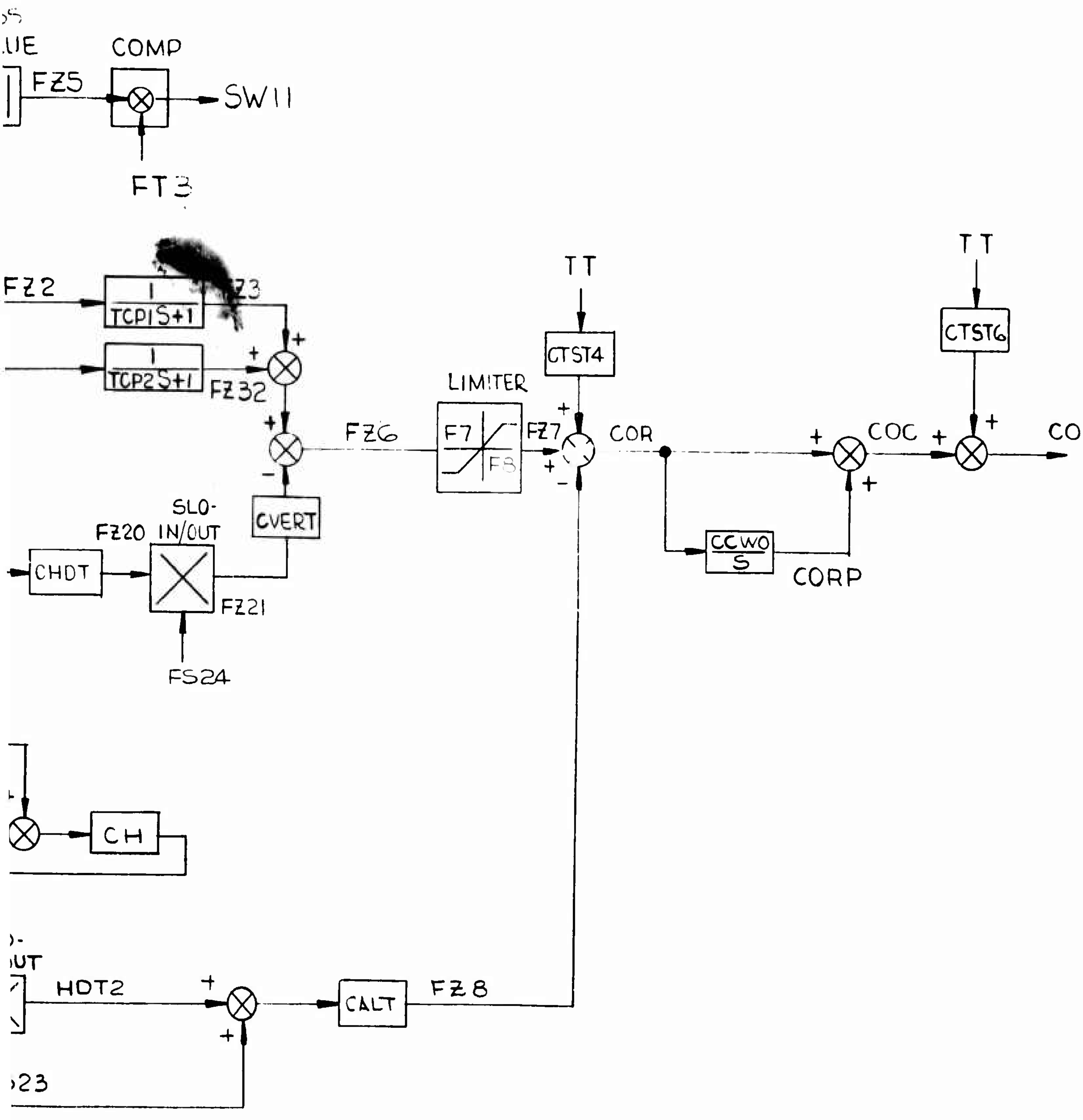


Figure 20. Collective Axis CSMP Diagram.

A


COMP
FZ5
SW11
FT3
FZ2
1
TCP1S+1
Z3
1
TCP2S+1
FZ32
FZ6
LIMITER
F7
F8
FZ7
TT
CTST4
COR
CVERT
SLO-
IN/OUT
FZ20
CHDT
FZ21
FS24
CCWO
S
CORP
COC
TT
CTST6
CO
CH
HDT2
CALT
FZ8

pilot force input in various modes and various aircraft operating conditions. Also, because the PAS is very non-linear, it is necessary to use this simulation in addition to the simplified longitudinal simulation described earlier.

This simulation has been set up to be used as a system test simulation. System responses with successive loop closures can be made in one pass on the digital computer. The successive loop closures have been set up to correspond to the technique that would be used for initial checking of the system in system simulation work or in actual flight testing. The computer results serve as a data reference base against which results of the aforementioned testing can be compared.

Complex Lateral CSMP Description

Block diagram of the roll and yaw axes of the complex lateral CSMP simulation are shown in Figures 21 and 22, respectively. Descriptions of the airframe 3-DOF equations of motion are shown on Table XII. Tables XIII through XV tabulate the aircraft stability derivatives for flight conditions 1, 4 and 12, respectively.

The rationale behind the lateral simulation is the same as that described for the complex longitudinal simulation. Except that Figure 22 does not reflect the simplified final yaw mechanization, but it can be easily modified .

Simplified 6-DOF CSMP Description

A block diagram of the simplified 6-DOF CSMP simulation is shown in Figure 23. Descriptions of the airframe 6-DOF equations of motion are shown in Table XVI. Tables XVII through XIX tabulate the aircraft stability derivatives for flight conditions 1, 4 and 12, respectively. Table XX shows a comparison of the cross-coupling derivatives for flight conditions 1, 4 and 12.

This simulation has been set up to check the effect of decoupling, by PAS feedbacks, the normal aircraft aerodynamic ccupling that exists. The simulation permits evaluation of the effect of servo bandwidth and also the effect of each decoupling term (either individually or in groups).

Minor Loop CSMP Description

A block diagram of the torque motor servo loop (minor loop) CSMP diagram is shown in Figure 24. This simulation has been set up to:

1. Use a data reference base for subsystem (servo loop) tests.

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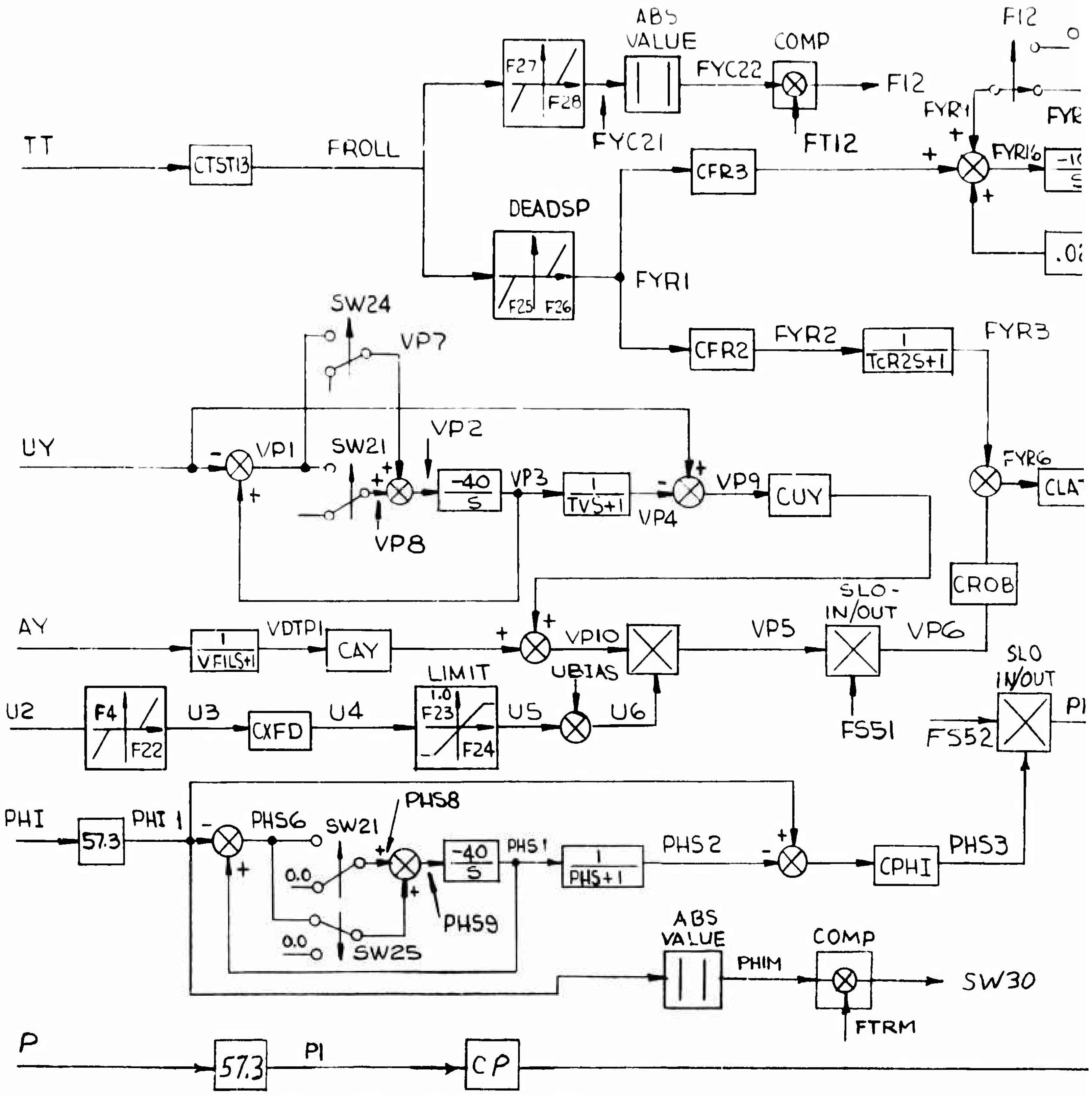
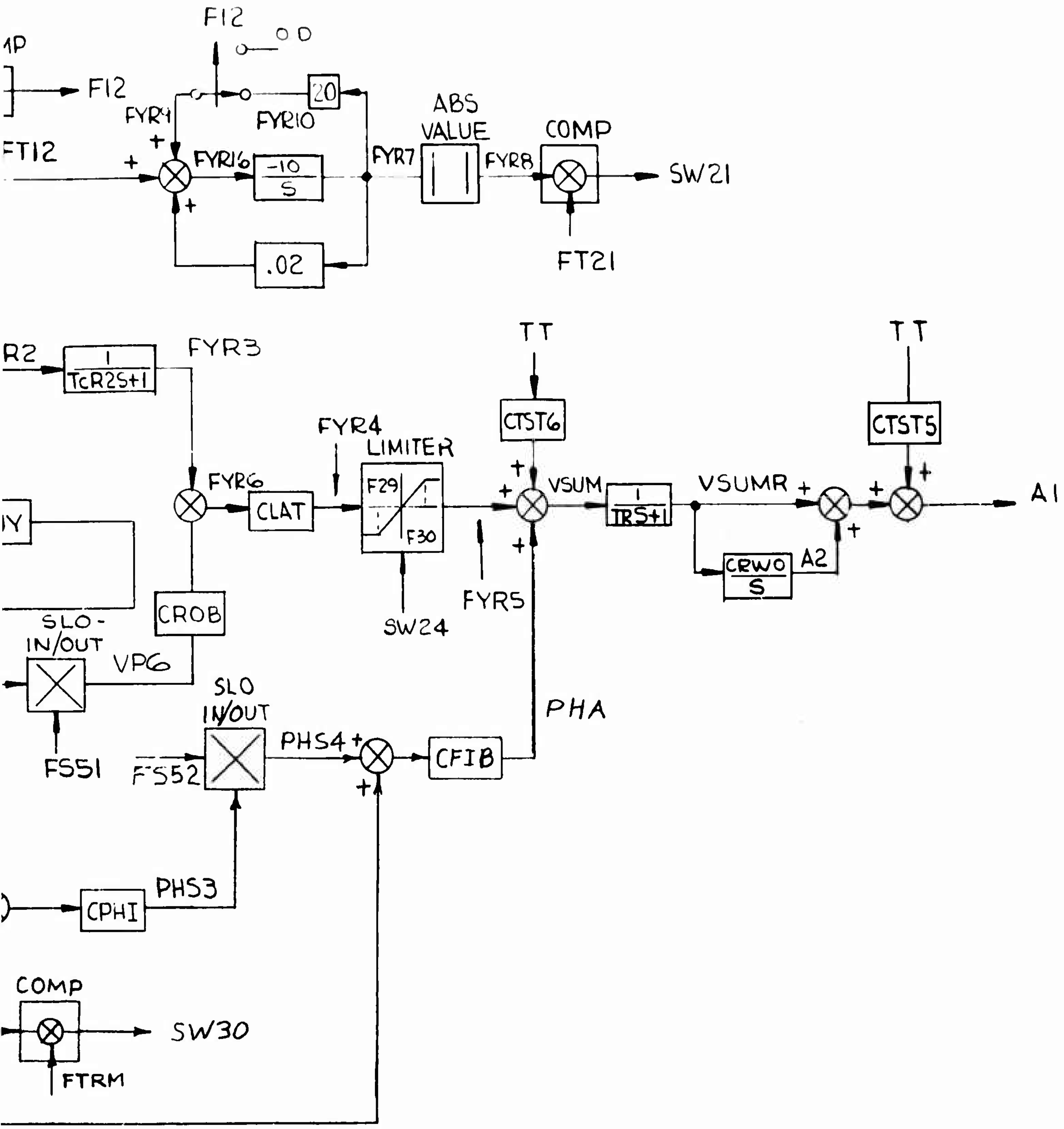


Figure 21. Roll Axis CSMP Diagram.

A



F12
F12
FYR9
FYR10
20
FT12
FYR16
-10/S
.02
FYR7
ABS VALUE
FYR8
COMP
SW21
FT21
R2
1/TcR2S+1
FYR3
TT
TT
CTST6
CTST5
FYR4
LIMITER
F29
F30
FYR6
CLAT
VSUM
1/TRS+1
VSUMR
A1
CRWO/S
A2
FYR5
SW24
CROB
SLO-IN/OUT
VP6
FS51
SLO IN/OUT
FS52
PHS4
CFIB
PHA
PHS3
CPHI
COMP
SW30
FTRM

B

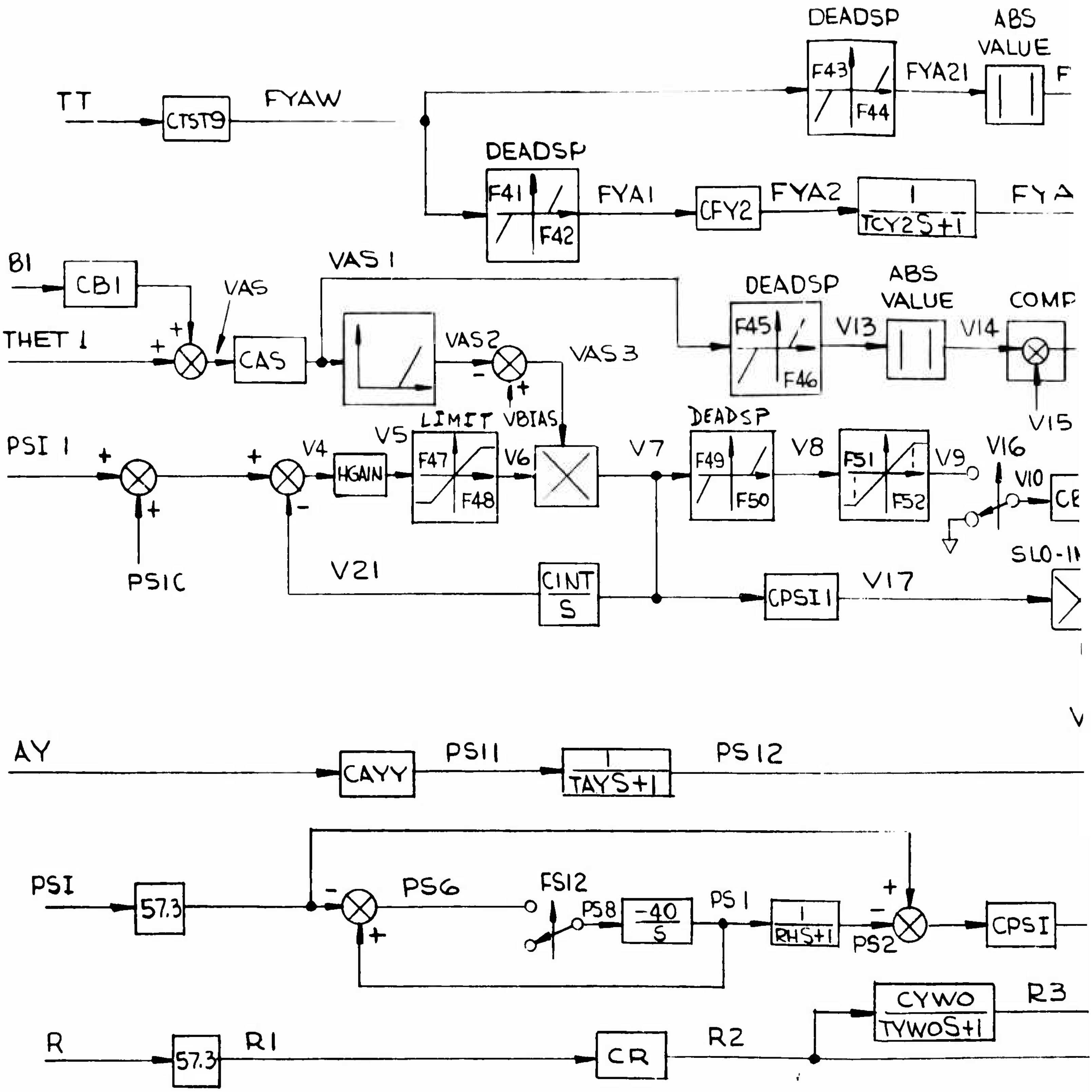
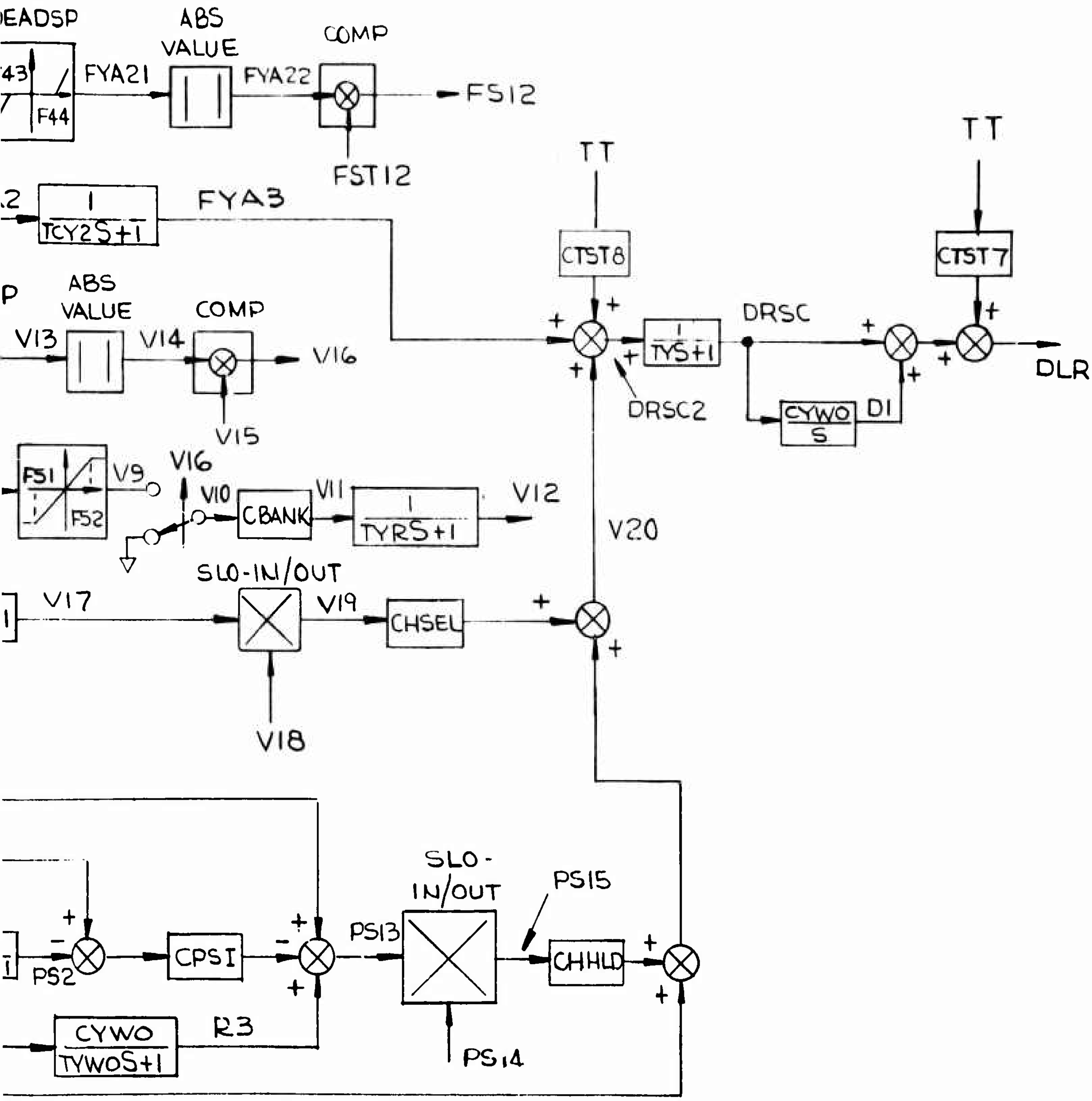


Figure 22. Yaw Axis CSMP Diagram.


DEADSP
F43
F44
FYA21
ABS
VALUE
FYA22
COMP
FS12
FST12
FYA3
1
TCY2S+1
TT
CTST8
1
TYS+1
DRSC
DRSC2
TT
CTST7
DLR
CYWO
S
D1
ABS
VALUE
V13
V14
COMP
V16
V15
FS1
F52
V9
V16
V10
CBANK
V11
1
TYRS+1
V12
V20
SLO-IN/OUT
V17
V19
CHSEL
V18
SLO-
IN/OUT
PS15
PS2
CPSI
PS13
CHHLD
PS14
CYWO
TYWOS+1
R3

B

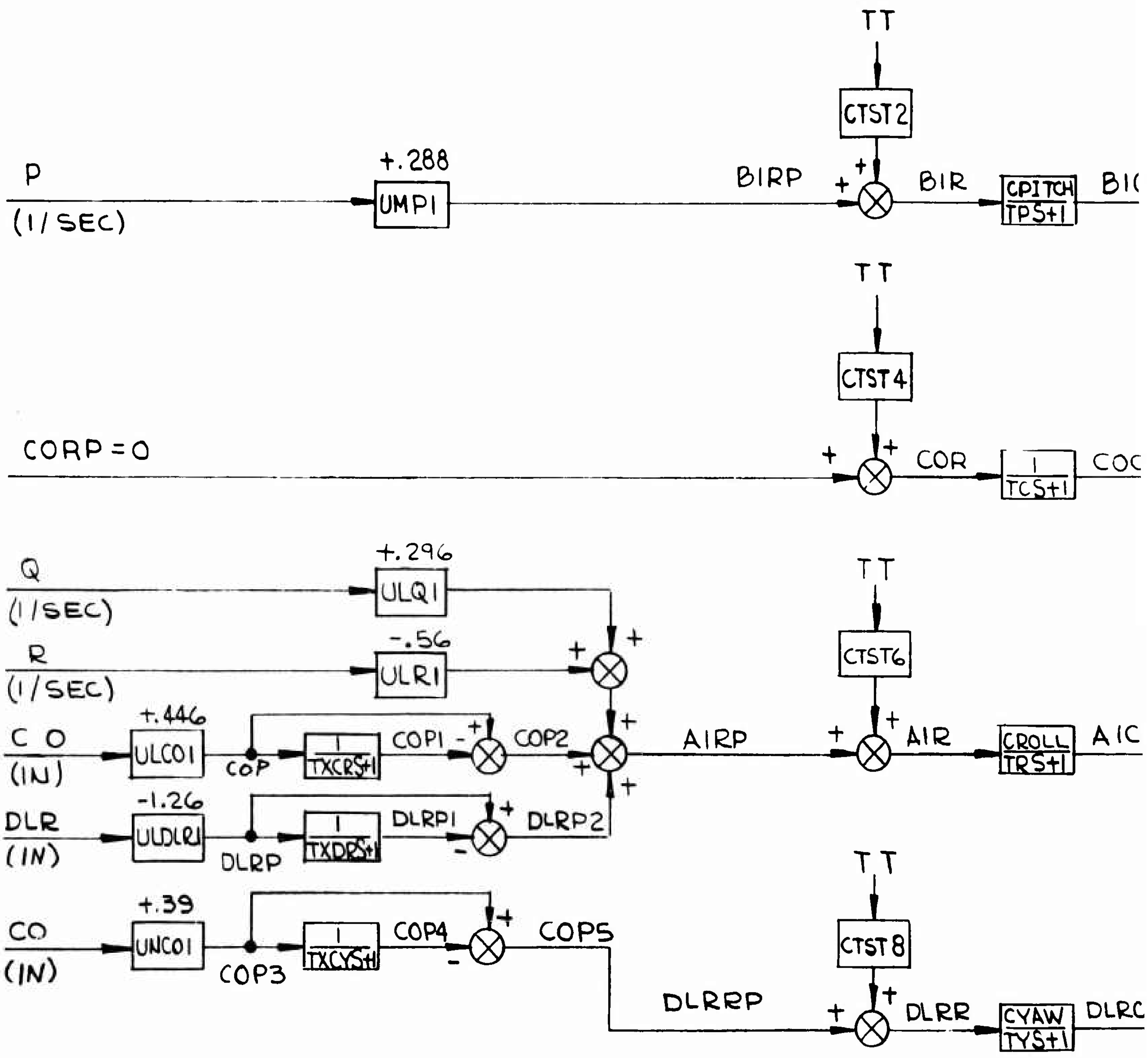
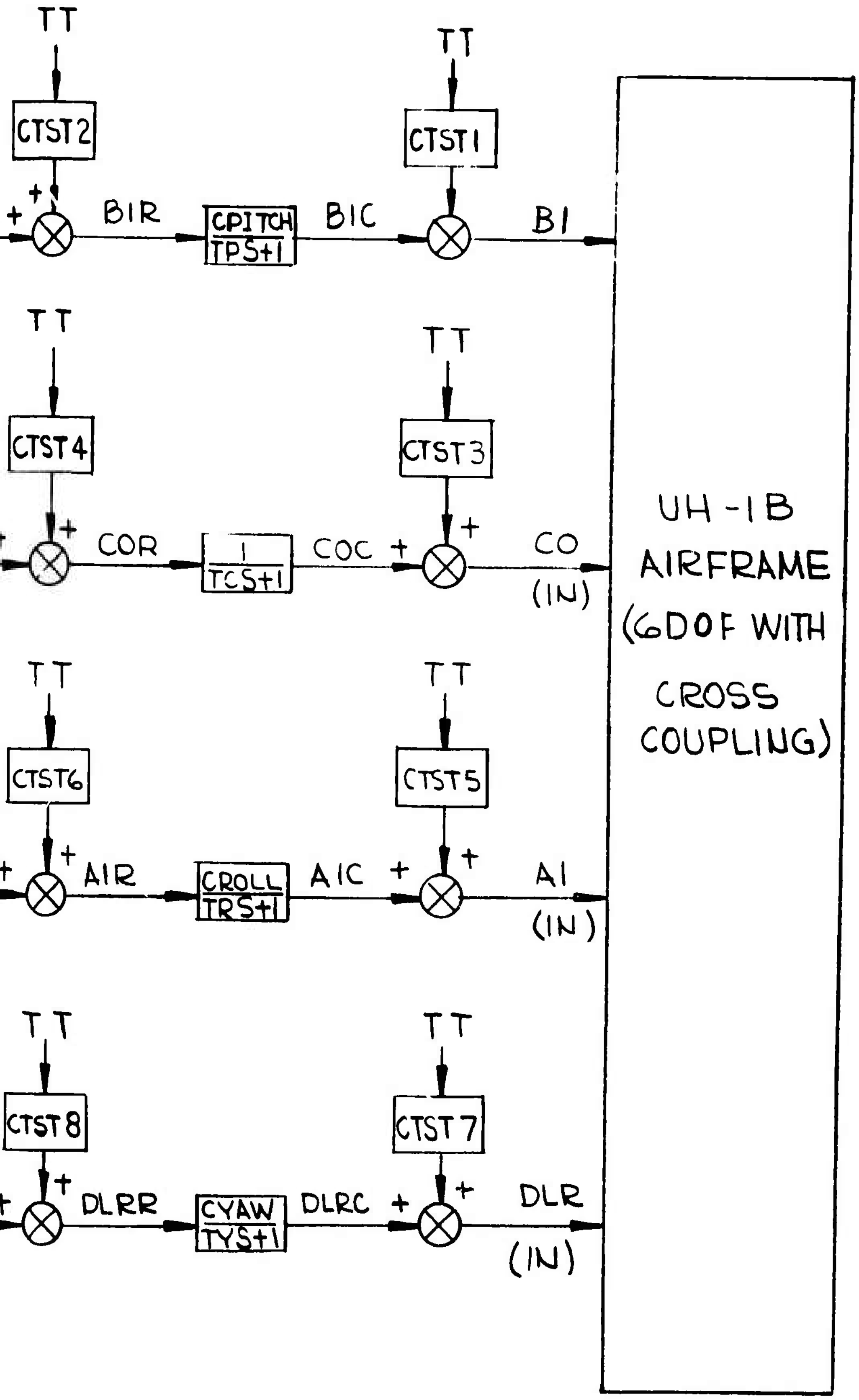


Figure 23. Simplified 6-DOF CSMP Simulation.


TT
CTST2
BIR
CPITCH
TPS+1
BIC
TT
CTST1
BI
TT
CTST4
COR
1
TCS+1
COC
TT
CTST3
CO
(IN)
UH-1B
AIRFRAME
(6DOF WITH
CROSS
COUPLING)
TT
CTST6
AIR
CROLL
TRS+1
AIC
TT
CTST5
AI
(IN)
TT
CTST8
DLRR
CYAW
TYS+1
DLRC
TT
CTST7
DLR
(IN)

B

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TABLE XVI. 6-DOF AIRCRAFT EQUATIONS OF MOTION

```
UDT=UX*U+XW*ALFA+UXQ*Q-GUO*THETA+UXB1*B1+UXCO*CO
+XV*BETA+UXP*P+UXR*R+UXA1*A1+UXDLR*DLR

ALFADT=ZU*U+ZW*ALFA+UZQ*Q-GUOS*THETA+UZB1*B1+UZCO*CO
+ZV*BETA+UZP*P+UZR*R+UZA1*A1+UZDLR*DLR

THE2DT=AMU*U+AMW*ALFA+UMQ*Q+UMB1*B1+UMCO*CO
+AMV*BETA+UMP*P+UMR*R+UMA1*A1+UMDLR*DLR

BETADT=YV*BETA+UYP*P+GUO*PHI+UYR*R+UYA1*A1+UYDLR*DLR
+YU*U+YW*ALFA+UYQ*Q+UYB1*B1+UYCO*CO

PHI2DP=ALV*BETA+ULP*P+UIXZ*RDTP+ULR*R+ULA1*A1+ULDLR*DLR
+ALU*U+ALW*ALFA+ULQ*Q+ULB1*B1+ULCO*CO

RDT=ANV*BETA+UNP*P+UIZZ*PHI2DT+ANR*R+UNA1*A1+UNDLR*DLR
+ANU*U+ANW*ALFA+ANQ*Q+UNB1*B1+UNCO*CO

PHI2DT=KIX*PHI2DP

RDTP=REALPL(0.0,TRDT,RDT)

BETA=INTGRL(0.0,BETADT)

PHIDT=INTGRL(0.0,PHI2DT)

P=PHIDT

R=INTGRL(0.0,RDT)

U=INTGRL(0.0,UDT)

ALFA=INTGRL(0.0,ALFADT)

THEDT=INTGRL(0.0,KIY*THE2DT)

THETA=INTGRL(0.0,THEDT)

Q=THEDT
```

TABLE XVII. 6-DOF AIRCRAFT STABILITY DERIVATIVES (FC1)

<u>Flight Condition 1</u>

U_o = 80 kt = 135.8 ft/sec

W = 6750 lb; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = -15.2 ft/sec

θ_o = -6.47 deg; $\sin \frac{\theta_o}{57.3} = -.113$

Alt = 3000 ft

C.G. = 134.4 in.

$XU=X_u/m$	$ZU=Z_u/m$	$AMU=U_oM_u/I_y$	$YU=Y_u/m$	$ALU=U_oL_u/I_x$	$ANU=U_oN_u/I_z$
= -.048	= -.0867	= -.377	= +.0007	= -.0248	= -.7674
$XV=X_v/m$	$ZV=Z_v/m$	$AMV=U_oM_v/I_y$	$YV=Y_v/m$	$ALV=U_oL_v/I_x$	$ANV=U_oN_v/I_z$
= +.0124	= -.0558	= +.0815	= -.5201	= -14.18	= +5.423
$XW=X_w/m$	$ZW=Z_w/m$	$AMW=U_oM_w/I_y$	$YW=Y_w/m$	$ALW=U_oL_w/I_x$	$ANW=U_oN_w/I_z$
= -.081	= -1.125	= +1.965	= -.0394	= -9.3702	= -1.1349

TABLE XVII - Continued

<u>Flight Condition 1 (Continued)</u>

$UXP=X_p/mU_o$ $= -.0039$	$UZP=Z_p/mU_o$ $= -.0170$	$UMP=M_p/I_y$ $= +.0902$	$UYP=Y_p/mU_o+\frac{W_o}{U_o}$ $= -.127$	$ULP=L_p/I_x$ $= -3.980$	$UNP=N_p/I_z$ $= -.0819$
$UXQ=X_q/mU_o-\frac{W_c}{U_o}$ $= +.122$	$UZQ=\left(1+\frac{Z_q}{mU_o}\right)$ $= +.986$	$UMQ=M_q/I_y$ $= -.513$	$UYQ=Y_q/mU_o$ $= -.0021$	$ULQ=L_q/I_x$ $= -.5421$	$ANQ=N_q/I_z$ $= +.2708$
$GUO=G/U_o$ $= +.237$	$GUOS=\frac{G}{U_o}\sin\left(\theta_d\right)$ $= -.027$	—	$GUO=G/U_o$ $= +.237$	$UIXZ=I_{xz}/I_x$ $= +1.411$	$UIZZ=I_{xz}/I_z$ $= +.1317$
$UXR=X_r/mU_o$ $= -.0004$	$UZR=Z_r/mU_o$ $= +.0015$	$UMR=M_r/I_y$ $= +.0197$	$UYR=\left(-1+\frac{Y_r}{mU_o}\right)$ $= -.985$	$ULR=L_r/I_x$ $= +2.856$	$ANR=N_r/I_z$ $= -1.388$
$UXB1=X_{B1}/mU_o$ $= +.012$	$UZB1=Z_{B1}/mU_o$ $= +.040$	$UMB1=M_{B1}/I_y$ $= -.313$	$UYB1=Y_{B1}/mU_o$ $= +.00176$	$ULB1=L_{B1}/I_x$ $= +.438$	$UNB1=N_{B1}/I_z$ $= +.0380$
$UXCO=X_{co}/mU_o$ $= -.011$	$UZCO=Z_{co}/mU_o$ $= -.161$	$UMCO=M_{co}/I_y$ $= +.320$	$UYCO=Y_{co}/mU_o$ $= -.0057$	$ULCO=L_{co}/I_x$ $= -1.36$	$UNCO=N_{co}/I_z$ $= +.537$
$UXA1=X_{A1}/mU_o$ $= -.00004$	$UZA1=Z_{A1}/mU_o$ $= -.00057$	$UMA1=M_{A1}/I_y$ $= -.0002$	$UYA1=Y_{A1}/mU_o$ $= +.00634$	$ULA1=L_{A1}/I_x$ $= +1.826$	$UNA1=N_{A1}/I_z$ $= -.0012$

TABLE XVII - Continued

Flight Condition 1 (Continued)

UXDLR $=X_{DLR}/mU_o$ $= -.00035$	UZDLR $=Z_{DLR}/mU_o$ $= +.00018$	UMDLR $=M_{DLR}/I_y$ $= +.0306$	UYDLR $=Y_{DLR}/mU_o$ $= +.0135$	ULDLR $=L_{DLR}/I_x$ $= +3.117$	UNDLR $=N_{DLR}/I_z$ $= -1.373$

TABLE XVIII. 6-DOF LONGITUDINAL STABILITY DERIVATIVES (FC4)

<u>Flight Condition 4</u>

U_o = 40 kt = 67.7 ft/sec

W = 6750 lb; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = -2.23 ft/sec

θ_o = -1.881 deg; $\sin \frac{\theta_o}{57.3} = -.033$

Alt = 3000 ft

C.G. = 134.4 in.

$XU=X_u/m$ = -.029	$ZU=Z_u/m$ = -.163	$AMU=U_oM_u/I_y$ = -.053	$YU=Y_u/m$ = +.007	$ALU=U_oL_u/I_x$ = +.765	$ANU=U_oN_u/I_z$ = -.594
$XV=X_v/m$ = +.0102	$ZV=Z_v/m$ = -.052	$AMV=U_oM_v/I_y$ = +.077	$YV=Y_v/m$ = -.274	$ALV=U_oL_v/I_x$ = -5.35	$ANV=U_oN_v/I_z$ = +1.839
$XW=X_w/m$ = -.018	$ZW=Z_w/m$ = -.900	$AMW=U_oM_w/I_y$ = +.183	$YW=Y_w/m$ = -.023	$ALW=U_oL_w/I_x$ = -2.5	$ANW=U_oN_w/I_z$ = -3.56
$UXP=X_p/mU_o$ = -.0067	$UZP=Z_p/mU_o$ = -.015	$UMP=M_p/I_y$ = +.0724	$UYP=Y_p/mU_o+\frac{W_o}{U_o}$ = -.027	$ULP=L_p/I_x$ = -3.703	$UNP=N_p/I_z$ = -.261

TABLE XVIII - Continued

Flight Condition 4 (Continued)

$UXQ=X_q/mU_o-\frac{W_o}{U_o}$ $= +.0228$	$UZQ=\left(1+\frac{Z_q}{mU_o}\right)$ $= +.999$	$UMQ=M_q/I_y$ $= -.385$	$UYQ=Y_q/mU_o$ $= -.006$	$ULQ=L_q/I_x$ $= -.79$	$ANQ=N_q/I_z$ $= +.18$
$GUO=G/U_o$ $= +.474$	$GUOS=\frac{G}{U_o}\sin(\theta_o)$ $= -.023$	—	$GUO=G/U_o$ $= +.474$	$UIXZ=I_{xz}/I_x$ $= +1.41$	$UIZZ=I_{xz}/I_z$ $= +.132$
$UXR=X_r/mU_o$ $= -.0009$	$UZR=Z_r/mU_o$ $= +.0006$	$UMR=M_r/I_y$ $= +.0103$	$UYR=\left(-1+\frac{Y_r}{mU_o}\right)$ $= -.980$	$ULR=L_r/I_x$ $= +1.993$	$ANR=N_r/I_z$ $= -.949$
$UXB1=X_{B1}/mU_o$ $= +.019$	$UZB1=Z_{B1}/mU_o$ $= +.033$	$UMB1=M_{B1}/I_y$ $= -.215$	$UYB1=Y_{B1}/mU_o$ $= +.0012$	$ULB1=L_{B1}/I_x$ $= +.143$	$UNB1=N_{B1}/I_z$ $= +.0167$
$UXCO=X_{co}/mU_o$ $= -.0069$	$UZCO=Z_{co}/mU_o$ $= -.260$	$UMCO=M_{co}/I_y$ $= +.075$	$UYCO=Y_{co}/mU_o$ $= -.007$	$ULCO=L_{co}/I_x$ $= -.784$	$UNCO=N_{co}/I_z$ $= +.568$
$UXA1=X_{A1}/mU_o$ $= 0$	$UZA1=Z_{A1}/mU_o$ $= -.0007$	$UMA1=M_{A1}/I_y$ $= 0$	$UYA1=Y_{A1}/mU_o$ $= +.0123$	$ULA1=L_{A1}/I_x$ $= +1.760$	$UNA1=N_{A1}/I_z$ $= -.00053$
UXDLR $= X_{DLR}/mU_o$ $= -.0006$	UZDLR $= Z_{DLR}/mU_o$ $= +.00007$	UMDLR $= M_{DLR}/I_y$ $= +.0144$	UYDLR $=Y_{DLR}/mU_o$ $= +.019$	ULDLR $= L_{DLR}/I_x$ $= +2.20$	UNDLR $= N_{DLR}/I_z$ $= -.9660$

TABLE XIX. 6-DOF AIRCRAFT STABILITY DERIVATIVES (FC12)

<u>Flight Condition 12</u>

U_o = 2 kt = 3.40 ft/sec

W = 6750 lb; m = 209.5 $\frac{\text{lb-sec}^2}{\text{ft}}$

I_x = 700 slug-ft^2; I_y = 9300 slug-ft^2

I_z = 7500 slug-ft^2; I_{xz} = 988 slug-ft^2

W_o = .000132 ft/sec

θ_o = +.219 deg; $\sin \frac{\theta_o}{57.3}$ = .000066

Alt = 3000 ft

C.G. = 134.4 in.

$XU=X_u/m$ = −.013	$ZU=Z_u/m$ = −.168	$AMU=U_oM_u/I_y$ = +.0057	$YU=Y_u/m$ = .0129	$ALU=U_oL_u/I_x$ = .038	$ANU=U_oN_u/I_z$ = −.013
$XV=X_v/m$ = .0042	$ZV=Z_v/m$ = −.107	$AMV=U_oM_v/I_y$ = .0048	$YV=Y_v/m$ = −.05	$ALV=U_oL_v/I_x$ = −.219	$ANV=U_oN_v/I_z$ = +.044
$XW=X_w/m$ = +.004	$ZW=Z_w/m$ = −.450	$AMW=U_oM_w/I_y$ = −.005	$YW=Y_w/m$ = −.0208	$ALW=U_oL_w/I_x$ = −.133	$ANW=U_oN_w/I_z$ = .00013
$UXP=X_p/mU_o$ = −.159	$UZP=Z_p/mU_o$ = −.046	$UMP=M_p/I_y$ = .0867	$UYP=Y_p/mU_o+\frac{W_o}{U_o}$ = −.207	$ULP=L_p/I_x$ = −1.44	$UNP=N_p/I_z$ = +.059

TABLE XIX - Continued

Flight Condition 12 (Continued)

$UXQ = X_q/mU_o - \frac{W_o}{U_o}$ $= +.172$	$UZQ = \left(1 + \frac{Z_q}{mU_o}\right)$ $= +1.213$	$UMQ = M_q/I_y$ $= -.108$	$UYQ = Y_q/mU_o$ $= -.148$	$ULQ = L_q/I_x$ $= -1.065$	$ANQ = N_q/I_z$ $= .00767$
$GUO = G/U_o$ $= +9.48$	$GUOS = \frac{G}{U_o} \sin(\theta_o)$ $= +.038$	—	$GUO = G/U_o$ $= +9.48$	$UIXZ = I_{xz}/I_x$ $= 1.41$	$UIZZ = I_{xz}/I_z$ $= +.132$
$UXR = X_r/mU_o$ $= -.013$	$UZR = Z_r/mU_o$ $= .0085$	$UMR = M_r/I_y$ $= .0075$	$UYR = \left(-1 + \frac{Y_r}{mU_o}\right)$ $= -.826$	$ULR = L_r/I_x$ $= +.996$	$ANR = N_r/I_z$ $= -.420$
$UXB1 = X_{B1}/mU_o$ $= +.370$	$UZB1 = Z_{B1}/mU_o$ $= +.0546$	$UMB1 = M_{B1}/I_y$ $= -.205$	$UYB1 = Y_{B1}/mU_o$ $= .003$	$ULB1 = L_{B1}/I_x$ $= .02$	$UNB1 = N_{B1}/I_z$ $= -.00013$
$UXCO = X_{co}/mU_o$ $= +.017$	$UZCO = Z_{co}/mU_o$ $= -4.71$	$UMCO = M_{co}/I_y$ $= -.009$	$UYCO = Y_{co}/mU_o$ $= -.271$	$ULCO = L_{co}/I_x$ $= -1.756$	$UNCO = N_{co}/I_z$ $= .861$
$UXA1 = X_{A1}/mU_o$ $= 0$	$UZA1 = Z_{A1}/mU_o$ $= -.014$	$UMA1 = M_{A1}/I_y$ $= 0$	$UYA1 = Y_{A1}/mU_o$ $= +.244$	$ULA1 = L_{A1}/I_x$ $= +1.76$	$UNA1 = N_{A1}/I_z$ $= -.0023$
UXDLR $= X_{DLR}/mU_o$ $= -.0014$	UZDLR $= Z_{DLR}/mU_o$ $= 0$	UMDLR $= M_{DLR}/I_y$ $= .0159$	UYDLR $= Y_{DLR}/mU_o$ $= +.384$	ULDLR $= L_{DLR}/I_x$ $= +2.22$	UNDLR $= N_{DLR}/I_z$ $= -.971$

TABLE XX. 6-DOF AERODYNAMIC CROSS-COUPLING TERMS

Term (CSMP Designation)	FC1 (80 kt)	FC4 (40 kt)	FC12 (2 kt)	Comments
1. UMP	+ .0902	+ .0724	+ .087	Use B1 = +.288 p
UMB1	- .313	- .215	- .205	
UMP/UMB1	- .288	- .337	- .425	
2. ULQ	- .542	- .79	-1.07	Use A1 = +.296 q
ULA1	+1.83	+1.76	+1.76	
ULQ/ULA1	- .296	- .45	- .61	
3. ULR	+2.86	+1.99	+ .996	Use A1 = -.56 r
ULA1	+1.83	+1.76	+1.76	
ULR/ULA1	+1.56	+1.13	+ .566	
4. ULCØ	-1.36	- .784	-1.76	Use A1 = +.466CØ' (through washout)
ULA1	+1.83	+1.76	+1.76	
ULCØ/ULA1	- .744	- .446	-1.0	
5. ULDLR	+3.12	+2.20	+2.22	Use A1 = -1.26 DLR' (through washout)
ULA1	+1.83	+1.76	+1.76	
ULDLR/ULA1	+1.71	+1.25	+1.26	
6. UNCØ	+ .54	+ .57	+ .861	Use DLR = +.39 CØ' (through washout)
UNDLR	-1.37	- .966	- .971	
UNCØ/UNDLR	- .394	- .59	- .887	

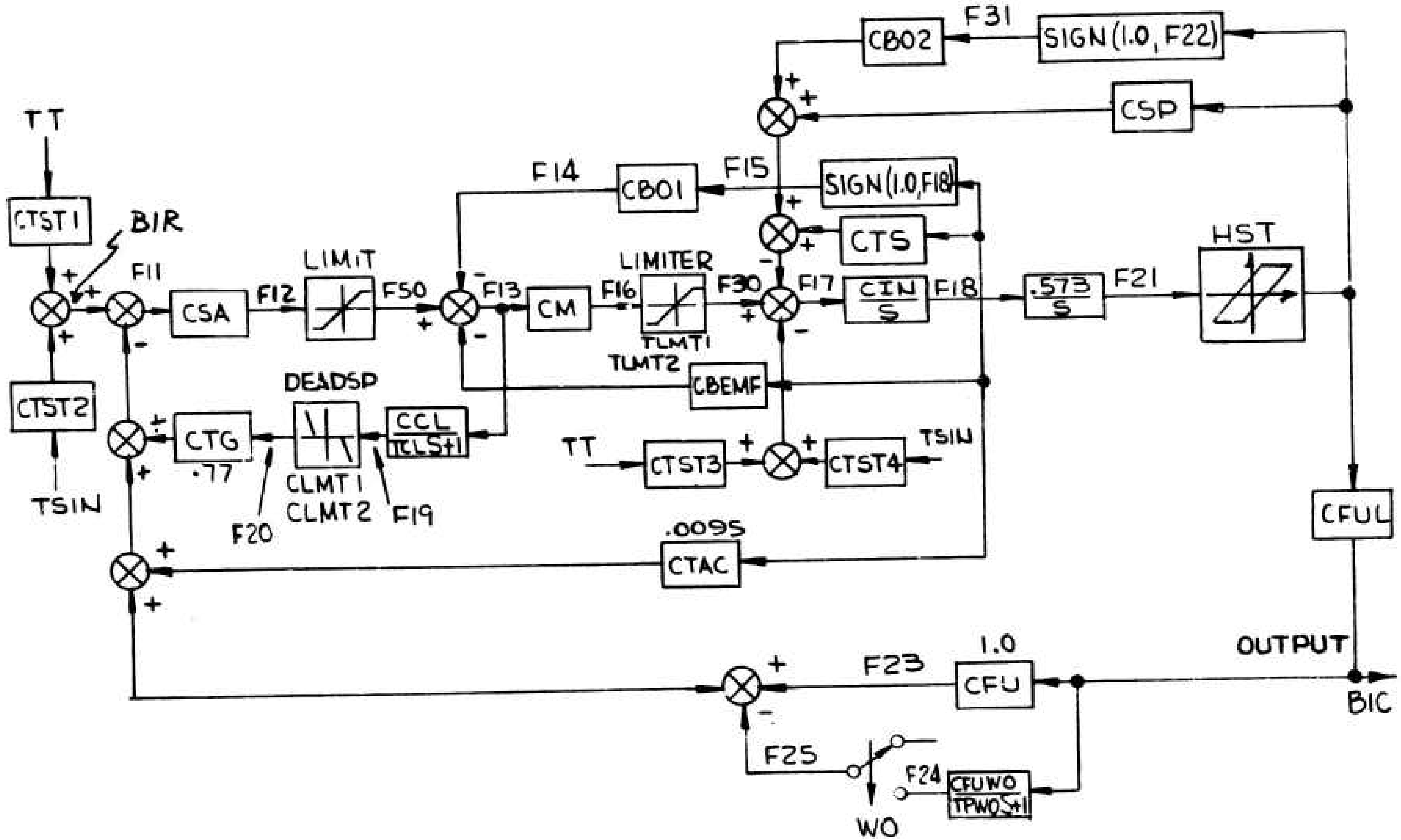

Figure 24. Minor Loop CSMP Diagram.

2. To have a more precise servo loop model available for inclusion into the other CSMP simulations, if future results prove that this is necessary.

Root Locus Digital Programs

Longitudinal Root Locus Program Description

Table XXI is a matrix diagram of the system described by the simplified longitudinal CSMP block diagram (Figure 12) and the 3-DOF longitudinal aircraft equations. This root locus program was set up to determine nominal system gains and to observe the trends that result when varying such parameters as:

1. Individual loop gains.

2. Individual loop time constants.

3. Servo bandwidth.

4. Servo washout time constant.

5. Accelerometer location.

The longitudinal root locus program is used in conjunction with the simplified longitudinal CSMP program to obtain the final design. This is done because of the difficulty in directly relating time response to pole-zero location in a complex higher order system.

The entries in Table XXI can be related to the corresponding block diagram and equations of motion by setting each row equal to zero and manipulating the resulting equation.

Longitudinal Root Locus Results

Table XXII contains a summary of the 6750-pound forward flight characteristic equations and the transfer functions for various forward speeds. The ANC results were generated using a root locus program. The ANC digital results are very close to the USAAVLABS digital results. It might be noted that the 80-knot flight condition contains a pole that is quite far out in the right-half plane (RHP). This RHP pole yields a fore/aft cyclic pulse response that appears to be quite different from the actual aircraft response (although this is difficult to assess due to a lack of aircraft data under the same conditions).

TABLE XXI. SIMPLIFIED LONGITUDINAL M

	U (1/1) 1	α (rad) 2	θ (rad) 3	$h_{A/C}$ (ft) 4	Bl (in.) 5	CO (in.) 6
1	$-X_u$ $+mS$	$-X_w$	W/U_o $\left(-\frac{X_q}{U_o}+\frac{W_o}{U_o}\right)S$		$-\frac{X_{B1}}{U_o}$	$-\frac{X_{CO}}{U_o}$
2	$-Z_u$	$-Z_w$ $+mS$	$\frac{W}{U_o}\sin\theta_o$ $-\left(m+\frac{Z_q}{U_o}\right)S$		$-\frac{Z_{B1}}{U_o}$	$-\frac{Z_{CO}}{U_o}$
3	$-M_u$	$-M_w$	$-\frac{M_q}{U_o}S$ $+\frac{I_y}{U_o}S^2$		$-\frac{M_{B1}}{U_o}$	$-\frac{M_{CO}}{U_o}$
4		$+U_o$	$-U_o$	$+S$		
5					$+S$	
6						$+S$
7						
8						
9	$+U_o(C_u)$ $+U_o(C_{\dot{u}})S$		$-57.3(C_\theta)$ $-57.3(C_q)S$			
10			$+CH2DT(ALX)S^2$	$+CH$ $+CHDT\ S$ $+CH2DTS^2$		

A

FIED LONGITUDINAL MATRIX

1 .)	CO (in.) 6	BIS (in.) 7	COB (in.) 8	BIS2 (in.) 9	COB2 (in.) 10
B1 o	$-\frac{X_{CO}}{U_o}$				
31 o	$-\frac{Z_{CO}}{U_o}$				
31 o	$-\frac{M_{CO}}{U_o}$				
		-CPWO -S			
	+S		-CCWO -S		
		+1 +TPS		-1	
			+1 +TCS		-1
				+1	
					+1

B

TABLE XXII. LONGITUDINAL FREE AIRCRAF

Flt Cond	Wt (lb)	Alt (K ft)	Fwd Vel (kt)	Lat Vel (kt)	Trans Fcn	USAAVLABS Results
1	6750	3	80	0	Long Den (Δ)	$3.03 \times 10^{6} (S-.734)(S+2.22)[(S+.110)^{2}+(.276)$
2			60			$4.04 \times 10^{6} (S-.448)(S+1.60)[(S+.167)^{2}+(.312)$
3			50			$4.852 \times 10^{6} (S-.352)(S+1.$[illegible]$98)[(S+.199)^{2}+(.30$
4			40			$6.065 \times 10^{6} (S-.263)(S+1.141)[(S+.216)^{2}+(.27$
5			30			$8.086 \times 10^{6} (S-.166)(S+.887)[(S+.205)^{2}+(.198$
6			20			$1.213 \times 10^{7} [(S+.525)^{2}+(.120)^{2}][(S-.0732)^{2}+($
7			15			$1.617 \times 10^{7} [(S+.500)^{2}+(.181)^{2}][(S-.107)^{2}+(.$
8			10			$2.426 \times 10^{7} [(S+.477)^{2}+(.191)^{2}][(S-.136)^{2}+(.$
9			5			$4.852 \times 10^{7} [(S+.459)^{2}+(.167)^{2}][(S-.159)^{2}+(.$
10			4			$6.065 \times 10^{7} [(S+.457)^{2}+(.160)^{2}][(S-.163)^{2}+(.$
11			3			$8.086 \times 10^{7} [(S+.455)^{2}+(.151)^{2}][(S-.167)^{2}+(.$
12			2			$1.213 \times 10^{8} [(S+.453)^{2}+(.137)^{2}][(S-.169)^{2}+(.$

Flight Condition 1 Transfer F

1. $\frac{U}{B1} = \frac{3.666 \times 10^{4} (S+.87)\left[(S+.124)^{2}+2.473^{2}\right]}{\Delta} \frac{1}{in.}$ 4. $\frac{\;}{C}$

2. $\frac{\alpha}{B1} = \frac{1.23 \times 10^{5} (S-7.01)\left[(S+.0486)^{2}+.186^{2}\right]}{\Delta} \frac{1}{in.}$ 5. $\frac{\;}{C}$

3. $\frac{\theta}{B1} = \frac{-9.26 \times 10^{5} (S+.0614)(S+.89)}{\Delta} \frac{1}{in.}$ 6. $\frac{\;}{C}$

A

INAL FREE AIRCRAFT TRANSFER FUNCTIONS

esults	ANC Results	
$)\left[(S+.110)^2+(.276)^2\right]$	$3.03\times10^6(S-.734)(S+2.22)\left[(S+.110)^2+(.276)^2\right]$	$\leftarrow$ $\zeta=.371$, $\omega=.298$
$)\left[(S+.167)^2+(.312)^2\right]$	$4.042\times10^6(S-.448)(S+1.66)\left[(S+.167)^2+(.312)^2\right]$	$\leftarrow$ $\zeta=.433$, $\omega=.354$
$98)\left[(S+.199)^2+(.307)^2\right]$	$4.85\times10^6(S-.352)(S+1.398)\left[(S+.199)^2+(.307)^2\right]$	$\leftarrow$ $\zeta=.593$, $\omega=.366$
$41)\left[(S+.216)^2+(.273)^2\right]$	$6.065\times10^6(S-.263)(S+1.41)\left[(S+.216)^2+(.273)^2\right]$	$\leftarrow$ $\zeta=.621$, $\omega=.348$
$7)\left[(S+.205)^2+(.198)^2\right]$	$8.086\times10^6(S-.166)(S+.887)\left[(S+.205)^2+(.198)^2\right]$	$\leftarrow$ $\zeta=.720$, $\omega=.285$
$20)^2\right]\left[(S-.0732)^2+(.238)^2\right]$	$1.213\times10^7\left[(S-.0732)^2+(.238)^2\right]\left[(S+.525)^2+(.120)^2\right]$	$\leftarrow$ $\zeta=.975$, $\omega=.539$
$31)^2\right]\left[(S-.107)^2+(.282)^2\right]$	$1.617\times10^7\left[(S-.1073)^2+(.282)^2\right]\left[(S+.500)^2+(.181)^2\right]$	$\leftarrow$ $\zeta=.940$, $\omega=.532$
$91)^2\right]\left[(S-.136)^2+(.311)^2\right]$	$2.425\times10^7\left[(S-.136)^2+(.311)^2\right]\left[(S+.477)^2+(.191)^2\right]$	$\leftarrow$ $\zeta=.928$, $\omega=.514$
$57)^2\right]\left[(S-.159)^2+(.311)^2\right]$	$4.85\times10^7\left[(S-.159)^2+(.331)^2\right]\left[(S+.459)^2+(.167)^2\right]$	$\leftarrow$ $\zeta=.94$, $\omega=.489$
$50)^2\right]\left[(S-.163)^2+(.334)^2\right]$	$6.065\times10^7\left[(S-.163)^2+(.334)^2\right]\left[(S+.457)^2+(.160)^2\right]$	$\leftarrow$ $\zeta=.944$, $\omega=.484$
$51)^2\right]\left[(S-.167)^2+(.337)^2\right]$	$8.086\times10^7\left[(S-.167)^2+(.337)^2\right]\left[(S+.455)^2+(.151)^2\right]$	$\leftarrow$ $\zeta=.949$, $\omega=.479$
$17)^2\right]\left[(S-.169)^2+(.339)^2\right]$	$1.212\times10^8\left[(S-.169)^2+(.339)^2\right]\left[(S+.453)^2+(.137)^2\right]$	$\leftarrow$ $\zeta=.957$, $\omega=.473$

lition 1 Transfer Functions

$\frac{1}{\text{n.}}$ 4. $$\frac{U}{CO}=\frac{3.11\times10^5(S+2.39)\left[(S-.306)^2+(.472)^2\right]}{\Delta}\ \frac{1}{\text{in.}}$$

$\frac{1}{\text{n.}}$ 5. $$\frac{\alpha}{CO}=\frac{-4.94\times10^5(S-.165)\left[(S-.498)^2+(.471)^2\right]}{\Delta}\ \frac{1}{\text{in.}}$$

6. $$\frac{\theta}{CO}=\frac{-1.07\times10^5(S+.215)(S+10.83)}{\Delta}\ \frac{1}{\text{in.}}$$

B

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Figure 25 shows the variation in closed loop-roots as the pitch rate gain, K_q, is varied. Figures 26 and 27 show the variation in system roots as the attitude gain, K_θ, and servo washout time constant are varied.

<u>Lateral Root Locus Program Description</u>

Table XXIII is a matrix diagram of the system described by the simplified lateral CSMP block diagram (Figure 14) and the 3-DOF lateral aircraft equations. The same comments made about the longitudinal root locus program apply to this program.

<u>Lateral Root Locus Program Results</u>

Table XXIV contains a summary of the 6750-pound forward flight characteristic equations and the transfer functions for varying forward speeds.

Figure 28 shows the variation in closed-loop roots as the yaw rate gain is varied. Figures 29 and 30 show the variation in system roots as the roll rate and roll attitude gains are varied.

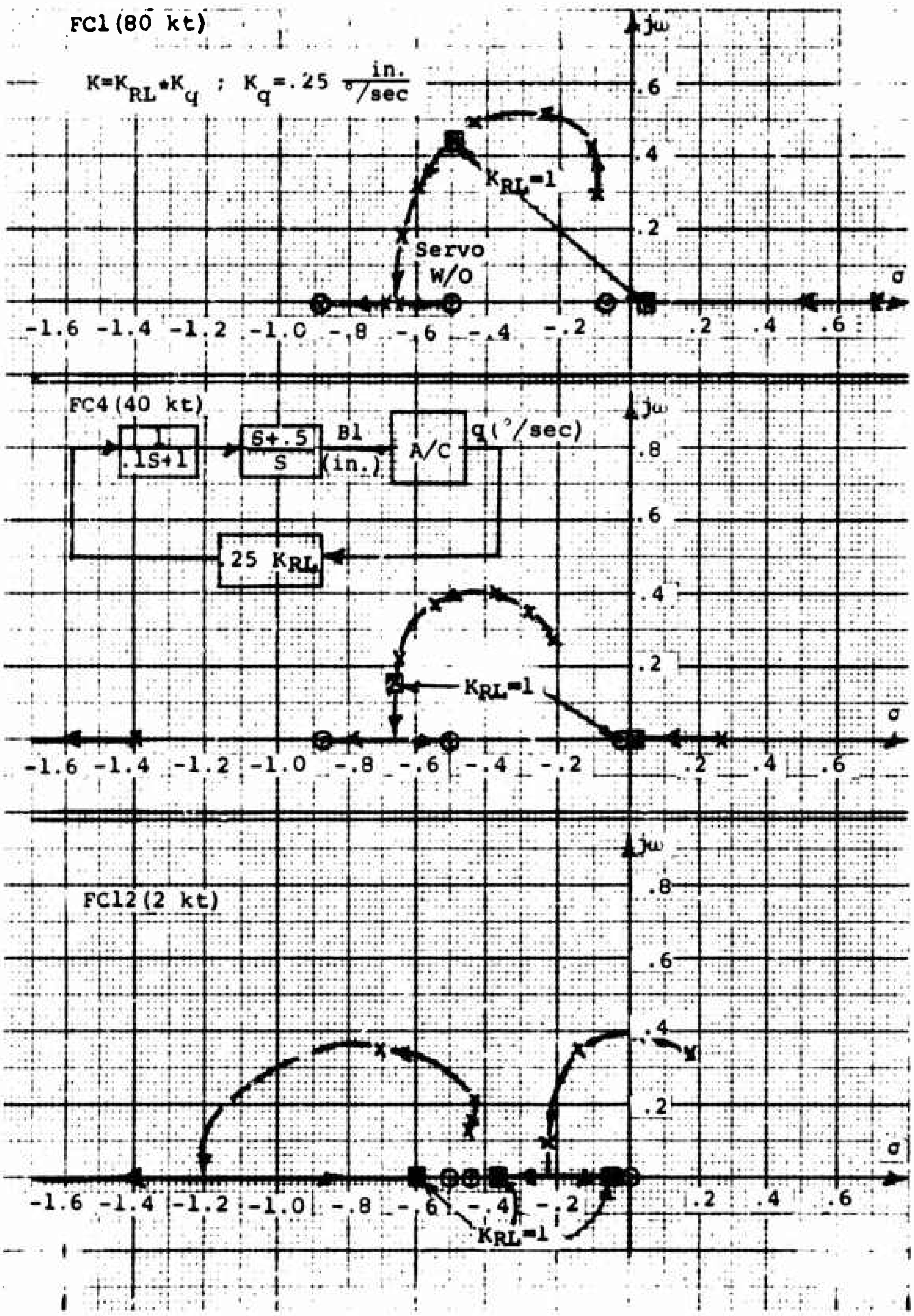

Figure 25. UH-1B Pitch Rate Root Locus.

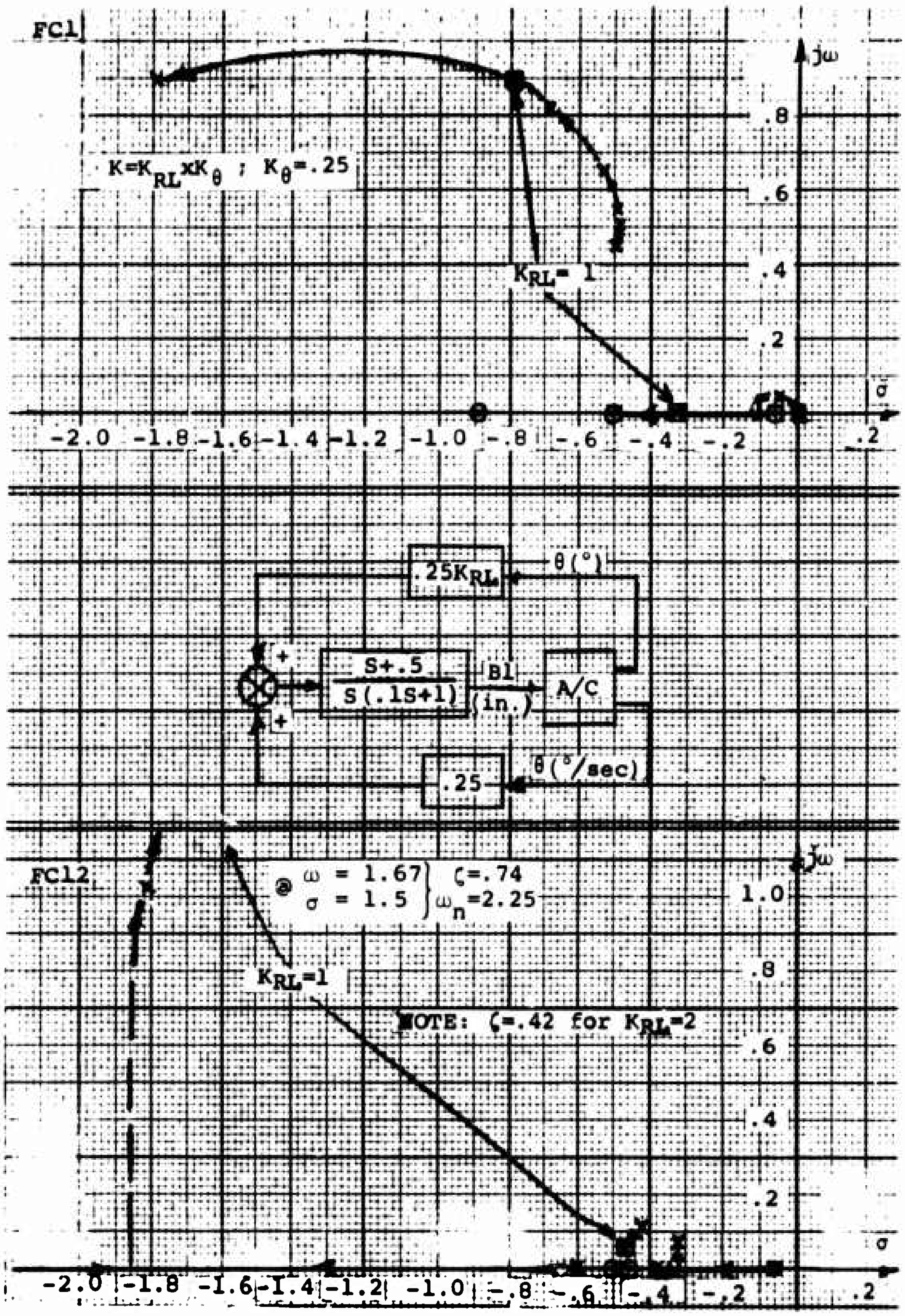

Figure 26. UH-1B Pitch Attitude Root Locus.

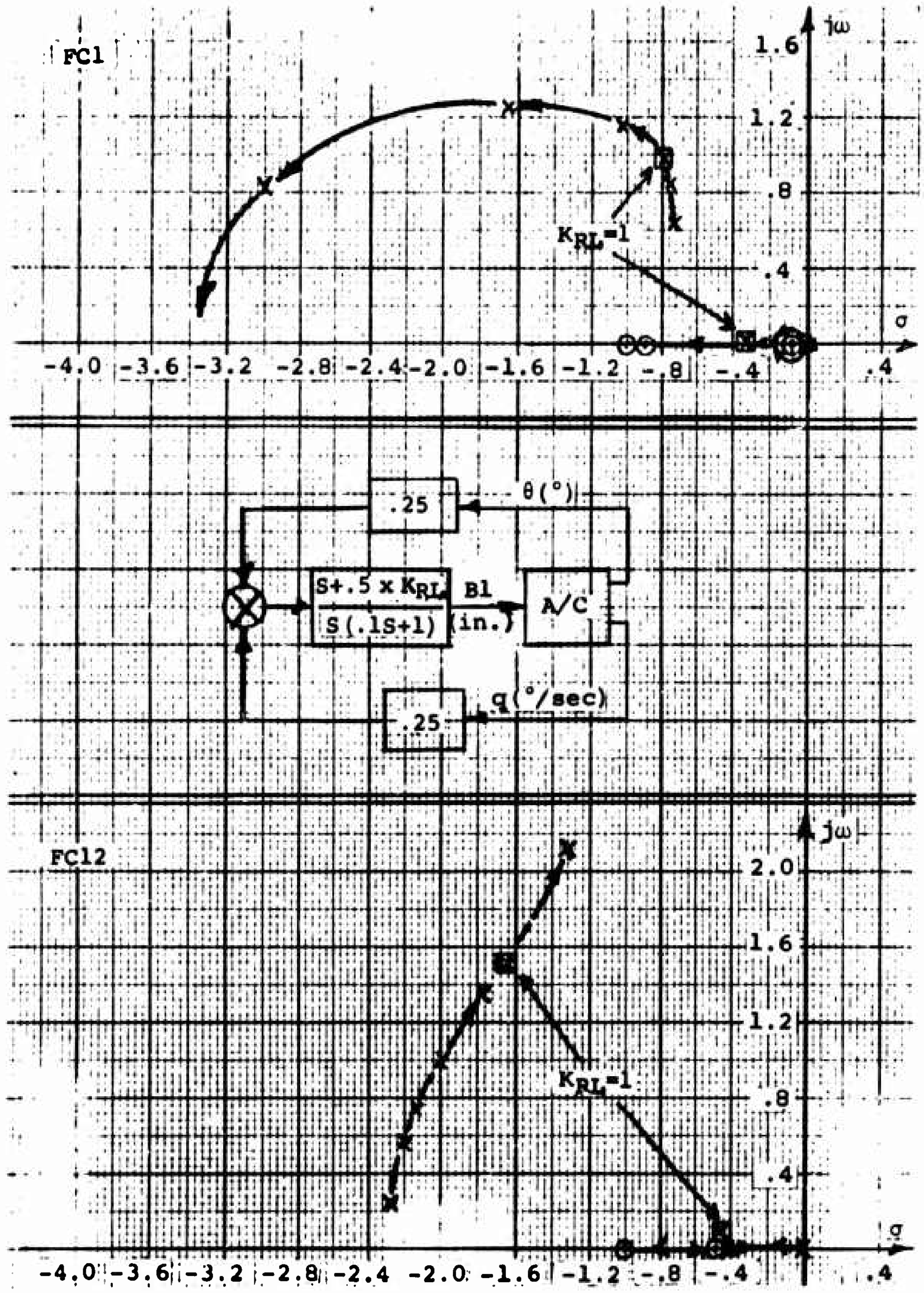

Figure 27. UH-1B Pitch Servo Washout Time Constant Root Locus.

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TABLE XXIII. SIMPLIFIED LATE

	β (rad)	φ (rad) 2	r (rad/sec) 3	$UY_{A/C}$ (ft/sec) 4	DLR (in.) 5	Al (in.) 6	DRSC (in. 7
1	$-Y_v$ $+mS$	$-W/U_o$ $-\left(\frac{Y_p}{U_o}+\frac{mW_o}{U_o}\right)S$	$m-\frac{Y_r}{U_o}$		$-\frac{Y_{DLR}}{U_o}$	$-\frac{Y_{Al}}{U_o}$	
2	$-L_v$	$-\frac{L_p}{U_o}S$ $+\frac{I_x}{U_o}S^2$	$-\frac{L_r}{U_o}$ $-\frac{I_{xz}}{U_o}S$		$-\frac{L_{DLR}}{U_o}$	$-\frac{L_{Al}}{U_o}$	
3	$-N_v$	$-\frac{N_p}{U_o}S$ $-\frac{I_{xz}}{U_o}S^2$	$-\frac{N_r}{U_o}$ $+\frac{I_z}{U_o}S$		$-\frac{N_{DLR}}{U_o}$	$-\frac{N_{Al}}{U_o}$	
4	$-\frac{U_oY_v}{m}$			S	$-\frac{Y_{DLR}}{m}$		
5					+S		-CYW(-S
6						+S	
7							+1 +TYS
8							
9			-57.3(CPSI)				
10		57.3(CPHI) 57.3(CP)S	-CAY(ALY)S	-CUY -CAY S			
11			-CAY(ALY)s	-CAYY S			
12			$-57.3C_rS$				

. SIMPLIFIED LATERAL MATRIX

A1 (in.) 6	DRSC (in.) 7	VSUMR (in.) 8	DRSC2 (in.) 9	VSUM (in.) 10	DRSC4 (in.) 11	R1P (deg) 12
$-\frac{Y_{A1}}{U_o}$						
$-\frac{L_{A1}}{U_o}$						
$-\frac{N_{A1}}{U_o}$						
	-CYWO -S					
+S		-CRWO -S				
	+1 +TYS		-1			
		+1 +TRS		-1		
			+S		-S	-S
				+1		
					1 +TAY S	
						YWO +S

B

TABLE XXIV. LATERAL FREE AIRCRAFT TRA

Flt Cond	Wt (lb)	Alt (K ft)	Fwd Vel (kt)	Lat Vel (kt)	Trans Fcn	USAAVLABS Results
1	6750	3	80	0	Lat Den (Δ)	$4.919x10^4 (S+.0374)(S+5.21)\left[(S+.770)^2+(2.37)^2\right]$
2			60			$8.744x10^4 (S+.0503)(S+5.48)\left[(S+.655)^2+(1.94)^2\right]$
3			50			$1.259x10^5 (S+.0497)(S+5.33)\left[(S+.583)^2+(1.72)^2\right]$
4			40			$1.967x10^5 (S+.0672)(S+5.06)\left[(S+.494)^2+(1.47)^2\right]$
5			30			$3.498x10^5 (S+.0958)(S+4.55)\left[(S+.379)^2+(1.22)^2\right]$
6			20			$7.870x10^5 (S+.159)(S+3.68)\left[(S+.220)^2+(1.01)^2\right]$
7			15			$1.399x10^6 (S+3.160)(S+.200)\left[(S+.130)^2+(.931)^2\right]$
8			10			$3.148x10^6 (S+.241)(S+2.672)\left[(S+.035)^2+(.885)^2\right]$
9			5			$1.260x10^7 (S+.273)(S+2.257)\left[(S-.070)^2+(.881)^2\right]$
10			4			$1.967x10^7 (S+.275)(S+2.180)\left[(S-.09)^2+(.89)^2\right]$
11			3			$3.498x10^7 (S+.279)(S+2.120)\left[(S-.119)^2+(.897)^2\right]$
12			2			$7.870x10^7 (S+.282)(S+2.070)\left[(S-.143)^2+(.905)^2\right]$

Flight Condition 1 Transfer Fu

1. $\frac{\beta}{Al} = \frac{3.14x10^2 (S+.782)(S-3.55)(S-41.51)}{\Delta} \frac{1}{in.}$

4. $\frac{\beta}{DLR} = 6.$

2. $\frac{\varphi}{Al} = \frac{1.102x10^5 \left[(S+.941)^2+(2.26)^2\right]}{\Delta} \frac{1}{in.}$

5. $\frac{\varphi}{DLR} = 7.$

3. $\frac{r}{Al} = \frac{1.45x10^4 (S+2.18)\left[(S-1.10)^2+(1.80)^2\right]}{\Delta} \frac{1/sec}{in.}$

6. $\frac{r}{DLR} = -$

A

EE AIRCRAFT TRANSFER FUNCTIONS

ults	ANC Results	
$.770)^2+(2.37)^2]$	$4.918\text{x}10^4(S+.0373)(S+5.21)\left[(S+.770)^2+(2.37)^2\right]$	← $\zeta=.309$ $\omega=2.49$
$.655)^2+(1.94)^2]$	$8.740\text{x}10^4(S+.0503)(S+5.48)\left[(S+.655)^2+(1.94)^2\right]$	← $\zeta=.320$ $\omega=2.046$
$.583)^2+(1.72)^2]$	$1.259\text{x}10^5(S+.0497)(S+5.33)\left[(S+.583)^2-(1.72)^2\right]$	← $\zeta=.321$ $\omega=1.82$
$.494)^2+(1.47)^2]$	$1.968\text{x}10^5(S+.067)(S+5.06)\left[(S+.494)^2+(1.47)^2\right]$	← $\zeta=.318$ $\omega=1.55$
$.379)^2+(1.22)^2]$	$3.498\text{x}10^5(S+.0956)(S+4.55)\left[(S+.379)^2+(1.225)^2\right]$	← $\zeta=.296$ $\omega=1.283$
$220)^2+(1.01)^2]$	$7.871\text{x}10^5(S+.159)(S+3.68)\left[(S+.220)^2+(1.01)^2\right]$	← $\zeta=.214$ $\omega=1.03$
$.130)^2+(.931)^2]$	$1.399\text{x}10^6(S+.200)(S+3.16)\left[(S+.130)^2+(.931)^2\right]$	← $\zeta=.138$ $\omega=.94$
$.035)^2+(.885)^2]$	$3.148\text{x}10^6(S+.241)(S+2.67)\left[(S+.035)^2+(.885)^2\right]$	← $\zeta=.04$ $\omega=.886$
$.070)^2+(.881)^2]$	$1.26\text{x}10^7(S+.273)(S+2.26)\left[(S-.07)^2+(.881)^2\right]$	← $\zeta=-.079$ $\omega=.884$
$.09)^2+(.89)^2]$	$1.968\text{x}10^7(S+.275)(S+2.18)\left[(S-.094)^2+(.888)^2\right]$	← $\zeta=-.105$ $\omega=.893$
$.119)^2+(.897)^2]$	$3.499\text{x}10^7(S+.279)(S+2.12)\left[(S-.119)^2+(.897)^2\right]$	← $\zeta=-.132$ $\omega=.905$
$.143)^2+(.905)^2]$	$7.871\text{x}10^7(S+.282)(S+2.07)\left[(S-.141)^2+(.905)^2\right]$	← $\zeta=-.154$ $\omega=.916$

1 Transfer Functions

4. $$\frac{\beta}{DLR} = \frac{6.714\text{x}10^2(S+.016)(S+6.43)(S+83.5)}{\Delta}\ \frac{1}{\text{in.}}$$

5. $$\frac{\varphi}{DLR} = \frac{7.13\text{x}10^4(S+1.86)(S-1.07)}{\Delta}\ \frac{1}{\text{in.}}$$

ec
. 6. $$\frac{r}{DLR} = \frac{-5.811\text{x}10^4(S+5.98)\left[(S+.218)^2+(.238)^2\right]}{\Delta}\ \frac{1/\text{sec}}{\text{in.}}$$

B

BLANK PAGE

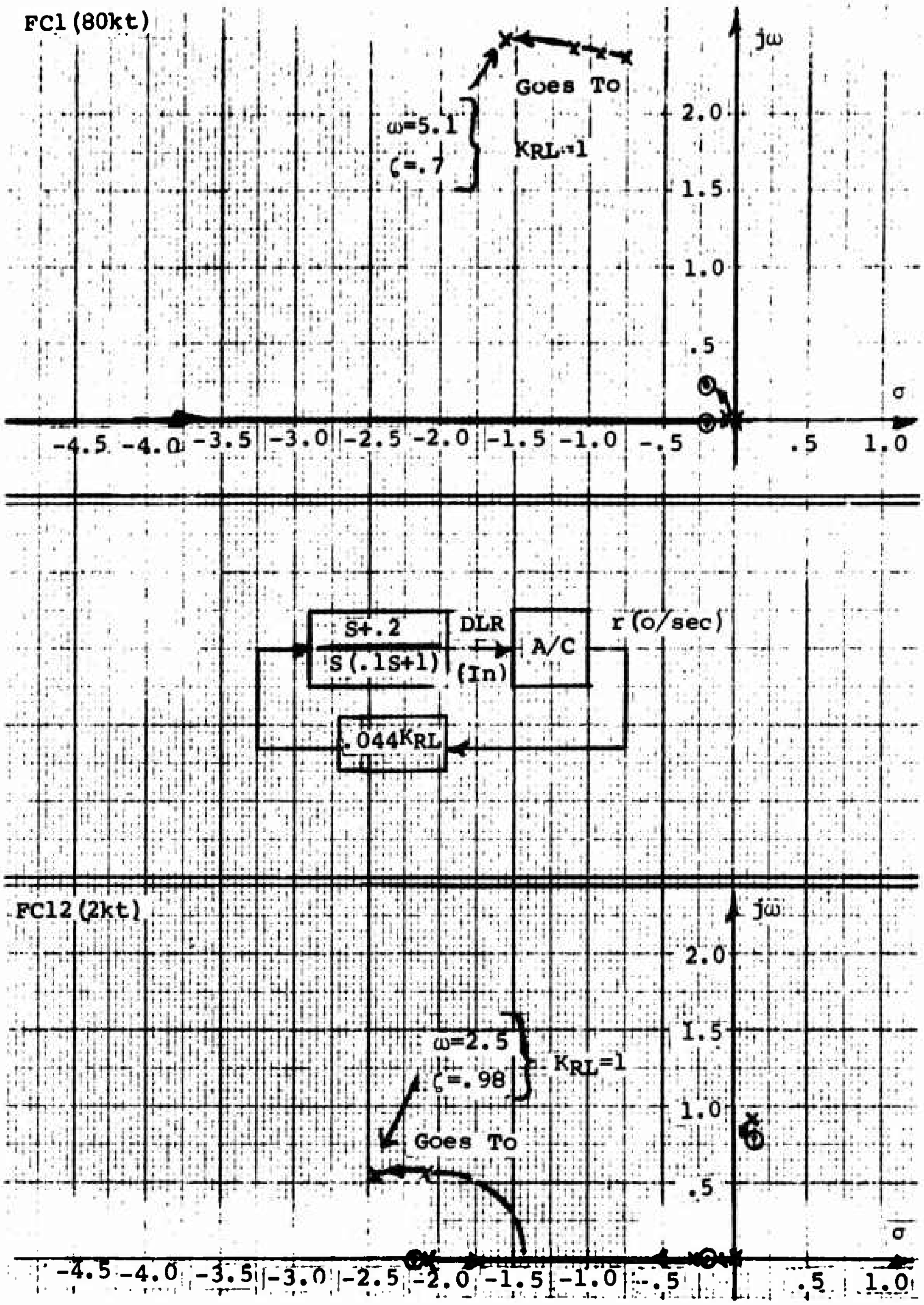

Figure 28. UH-1B Yaw Rate Root Locus.

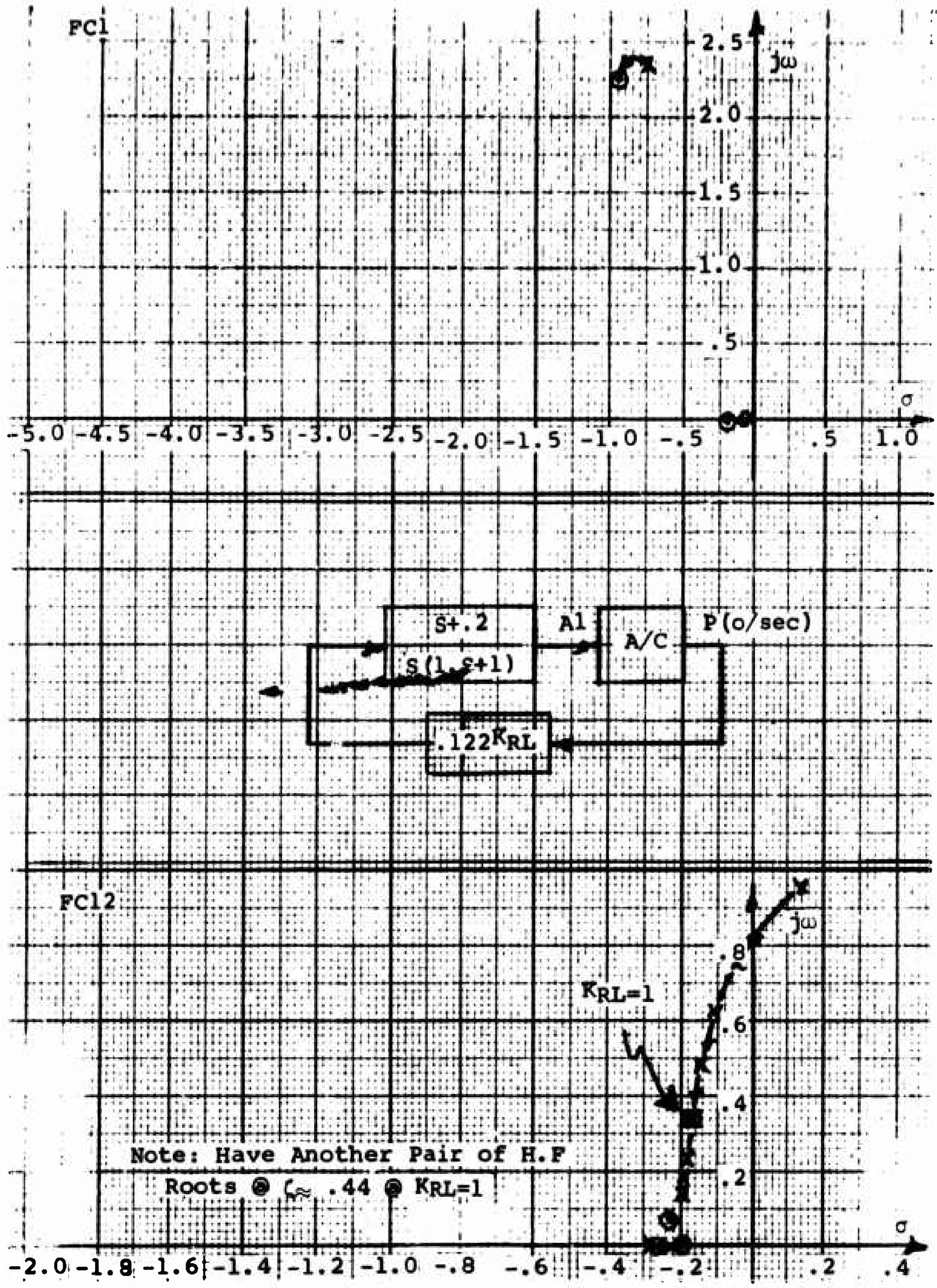

Figure 29. UH-1B Roll Rate Root Locus.

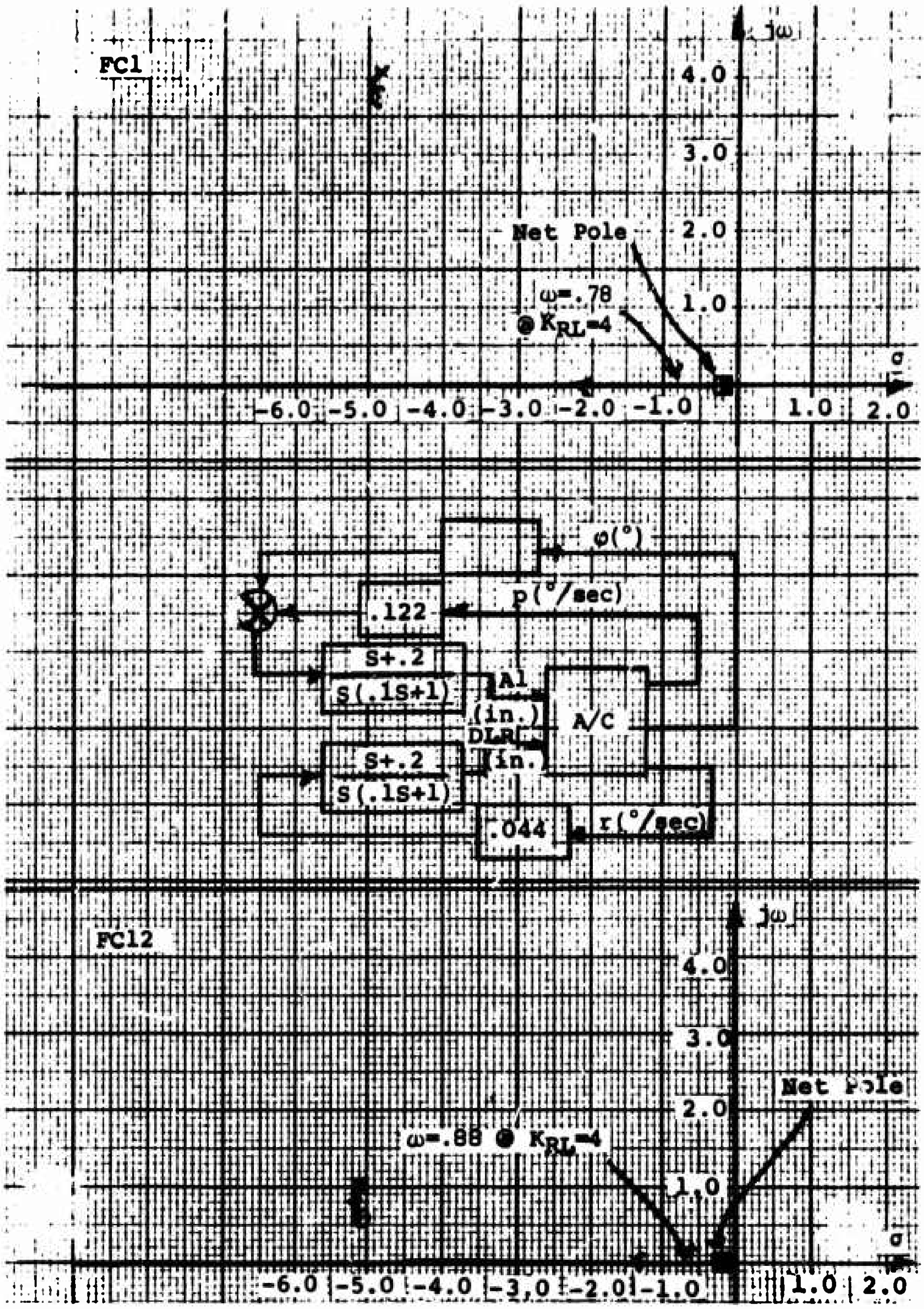

Figure 30. UH-1B Roll Attitude Root Locus.

UH-1B PAS DESIGN

SYSTEM DESCRIPTION

The pilot assist system (PAS) consists of the following ANC designed major subsystems:

1. Major Loop Computer (ANC Model No. R119) - one unit
2. Minor Loop Computer (ANC Model No. R120) - one unit
3. Mode Selector (ANC Model No. R121) - one unit
4. Cyclic Force Sensor (ANC Model No. FS-132) - two units
5. Collective Force Sensor With Friction Lock (ANC Model No. FS-133) - one unit
6. Collective Force Sensor Without Friction Lock (ANC Model No. FS-134) - one unit
7. Pedal Force Sensor (ANC Model No. FS-110) - one unit
8. Parallel Servo (ANC Model No. SA-104) - one unit
9. Series/Parallel Servo (ANC Model No. SA-103 - three units

A complete PAS-equipped flight test vehicle requires the following additional subsystems:

1. Government-furnished aircraft motion sensors
2. Equipment installation bracketry
3. An instrumentation package

Detailed descriptions of the above subsystems (1 through 9) have been generated as part of this work. This documentation, however, is delivered separately from this report.

Some of the general PAS characteristics are as follows:

1. The present prototype design is lightweight ($\approx$ 55 pounds), however, the weight will be lower for a production configuration (due to elimination of extra components now used to provide flexibility, testing convenience, etc.).

2. The PAS will allow the pilot to become more mission oriented by operating through an augmented aircraft.

3. The PAS will alleviate pilot workload by providing:

   a. Basic vehicle stability augmentation

   b. A reduction in the effect of basic vehicle cross-coupling

   c. Assistance in hover control

   d. Automatic cruise control

   e. Ease of maneuvering, including turn coordination

   f. Gust alleviation

Some of the features of the PAS, which have been provided for ease of further testing and development, are:

1. The capability to easily adjust (by accessible dial pots) all major loop computer leg gains and the minor loop computer tachometer gain and follow-up washout time constant.

2. The capability to adjust (via accessible trim pots) all other gains and time constants of interest.

3. The ability to easily compare the relative merits of series, series/parallel and parallel servo configurations via being able to quickly switch from one configuration to another.

4. Pilot force and servo command test inputs are provided for easily checking subsystem characteristics (e.g., major loop computer switching or minor loop closed-loop response) or system characteristics (e.g., applying a repeatable test pulse in the air or using an aircraft simulation and applying a test pulse on the ground).

5. The major loop computer dial pots have been so placed and scaled to have a wide range of adjustment (i.e., with a nominal of approximately .2 so that there is the capability to go a factor of 5 in each direction) and also not cause leg saturation at the high end. System gains can also be directly ascertained by reading the dial pot settings.

6. An extender board is provided for individually checking

printed circuit boards in both the major and minor loop computers.

The following test aids have been developed by ANC to aid in efficiently checking a bulk of the PAS characteristics:

1. A circuit board tester to test major and minor loop computer boards in the lab.

2. A test harness, simulated signal source, and servo load stand for testing the major and minor loop computers and for testing the closed-loop minor loop characteristics in the lab.

3. A portable analog simulation of the aircraft for use in testing the total system either in the lab or with the PAS installed in the aircraft.

MAJOR LOOP COMPUTER

Block diagrams that describe the contents of the major loop computer (R119) are shown in the axes diagrams of Figures 1, 3, 4 and 5. These diagrams are a near one-to-one math model representation of the major loop computer electronics.

Some of the design characteristics of the major loop computer are:

1. Printed circuit boards (both single and double sided) contain one type of integrated circuit function (e.g., filter) per board. The number of circuits per board varies between board types. There are 13 types of circuits in the major loop computer.

2. Each control path leg contains a dial pot for ease of setting and visually checking path gains.

3. Each control path leg contains balance circuitry for trimming each leg separately.

4. Each control path output (before it sums with another leg) is brought to a test connector for ease of monitoring and testing.

5. Slo-in circuitry is used to minimize mode switching transients.

MINOR LOOP

Figure 11 is a block diagram which describes the characteristics of each minor loop. From Figure 11 we see that the minor loop

consists of a torque motor servo (containing a follow-up synchro and a tachometer) and control electronics. The PAS contains four parallel torque motor servos, which have a rotary motion output, and three series torque motor servos, which have a linear motion output. The minor loop computer, therefore, contains seven sets of servo control circuitry to position the seven torque motor servos.

The characteristics of the minor loop control circuitry are as follows:

1. Each set of control circuitry is contained on two printed circuit boards (servo driver board and feedback board).

2. The servo driver boards contain a summing amplifier (with balance or trim capability) and two stages of power amplification.

3. The bulk of the electrical servo loop amplification has been placed in the last stage of amplification before the motor coils to minimize potential saturation problems.

4. The feedback circuit board contains current limiting and follow-up shaping circuitry.

MODE SELECTOR

A drawing of the mode selector panel layout is shown in Figure 2. The mode selector contains the switches and switching logic necessary to perform the mode engagement and switching indicated in the axes block diagrams (Figures 1, 3, 4 and 5). The mode selector also contains the switching necessary to select the desired servo test configuration in the cyclic axes and in yaw. The four switches in the top row are for individual engagement of the four control axes.

Many of the switch functions provided in this prototype design are for system mode evaluation purposes and would not be required in a final system design. For instance, the servo select switches in the bottom row are for the purpose of evaluating the three possible servo configurations in simulation and flight test using this prototype system design. Once the most desirable servo configuration is selected for each axis, these switches would serve no value and would be removed from the design. The same is true of the axis mode switches such as "ATT"/"VEL" or "HDG HLD"/"HDG SEL". They are provided in this prototype system design in order to allow evaluation of the various possible system modes easily and quickly during simulation and flight test. Once the desired modes are

selected, the unnecessary switches can be removed from the system design.

FORCE SENSORS

The force sensors provide electrical signals proportional to the forces applied by the pilot to each of his primary control devices, i.e., cyclic stick, collective stick and yaw pedals. The design philosophy followed in this program was to make the force sensors as unobtrusive as possible. Also, in the cases of the cyclic and collective stick force sensors, it was desired to place the sensor as close as possible to the pilot's grip for system dynamics reasons. The force sensor designs that have been developed as a result of this program have the following desirable characteristics:

1. They do not affect the characteristics of the basic aircraft control device.

2. They provide electrical signals which are directly usable in the pilot assist system major loop.

3. The design reduces the control stick "bob weight" effect, which tends to introduce undesired forces into the sensor, so it is below force loop thresholds.

CIRCUITS

The basic philosophy followed in the design of circuits and circuit boards for the pilot assist system was to develop the required circuits in functional groupings, i.e., summing amplifiers, filters, washouts, demodulators, synchronizers, etc. During the program, each of these circuits was breadboarded and individually tested. Then one axis of the pilot assist system was mechanized on a breadboard using these circuits for final development testing in an operating control loop. The final design resulted in 17 different circuit types, including power supplies.

With the functional circuits developed and checked to this degree, the layout of printed circuit boards for the prototype pilot assist system could begin. In the interest of flexibility, it was decided to maintain the functional circuit breakout, i.e., a summing amplifier board, a filter board, a washout board, a demodulator board, a synchronizer board, etc. Each circuit board type was designed to contain the maximum possible number of its circuit function. This approach results in an extremely flexible design which can be tailored to a wide variety of control system applications with a minimum of design change.

CONCLUSIONS

Work performed under Contract DAAJ02-70-C-0019 has resulted in a PAS hardware design which can be easily fabricated.

Subsequent system evaluation and refinement efforts should be minimized by the PAS design. A sizeable step has been taken toward conducting a ground-based simulation and/or flight test PAS evaluation.

RECOMMENDATIONS

ANC's recommendations for further work that will provide fruitful continuation of the work conducted under Contract DAAJ02-70-C-0019 are:

1. Test efforts to verify aircraft compatibility and sensor characteristics with the PAS.

2. The use of parallel flight test and ground based simulation efforts to provide an efficient evaluation and refinement of the PAS. The simulation effort should provide a data reference for verification of the flight test effort.

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5. Walton, R. P., and I. L. Ashkenas, ANALYTICAL REVIEW OF MILITARY HELICOPTER FLYING QUALITIES, August 1967, AD 666 827.

6. Users Manual, SYSTEM/360 CONTINUOUS SYSTEM MODELING PROGRAM, H20-0367-3, October 1969.

APPENDIX
DIGITAL COMPUTER PROGRAMS

This Appendix contains program listings for the following digital computer programs:

1. Simplified Longitudinal CSMP
2. Simplified Lateral CSMP
3. Complex Longitudinal CSMP
4. Simplified 6-DOF CSMP
5. Minor Loop CSMP

Also included is a typical program execution (Minor Loop CSMP run).

TABLE XXV. SIMPLIFIED LONGITUDINAL CSMP PROGRAM LISTING

```
             ****CONTINUOUS SYSTEM MODELING PROGRAM****

      ***PROBLEM INPUT STATEMENTS***

TITLE      JOB 72  SIMPLIFIED LONG CSMP LONG1
*       AIRCRAFT EQUATIONS (LONG)
        UDT      =XU*U+XW*ALFA+UXQ*Q-GUO*THETA+UXB1*B1+UXCO*CO ...
        +XU*UG+XW*ALFAG
        ALFADT   =ZU*U+ZW*ALFA+UZQ*Q-GUOS*THETA+UZB1*B1+UZCO*CO ...
        +ZU*UG+ZW*ALFAG
        THE2DT   =AMU*U+AMW*ALFA+UMQ*Q+UMB1*B1+UMCO*CO ...
        +AMU*UG+AMW*ALFAG
         U       =INTGRL(0.0,UDT)
        ALFA     =INTGRL(0.0,ALFADT)
        THEDT    =INTGRL(0.0,KIY*THE2DT)
        THETA    =INTGRL(0.0,THEDT)
        Q        =THEDT
        CAZ      =U1*(Q-ALFADT)+ALX*THE2DT
        HDT      =U1*(TH[illegible]-ALFA)+HDTIC
        H        =INTGRL [illegible]0,HDT)
        UDT1=UO*UDT
        U1=UO*(1.0+U)
        U2=UO*U
        Q1       =57.3*Q
        THET1    =57.3*THETA
        X=INTGRL(0.0,U)
PARAMETER    KIY=.800
PARAMETER    XU=-.048,XW=-.081,UXQ=+.122,GUO=+.237
PARAMETER    UXB1=+.012,UXCO=-.01
PARAMETER    ZU=-.087,ZW=-1.13,UZQ=+.986,GUOS=-.027
PARAMETER    UZB1=+.04,UZCO=-.161
PARAMETER    AMU=-.377,AMW=1.965,UMQ=-.513
PARAMETER    UMB1=-.313,UMCO=+.320
PARAMETER   HDTIC=0.0,UO=135.8,ALX=0.0
*
*                GUST INPUT
*
        NOISE=GAUSS(1,0.0,STD)
        FLTNS=REALPL(0.0,TNS,NOISE)
PARAMETER   STD=5.0,TNS=3.18
        ALFAG=ANS1*FLTNS
        UG=ANS2*FLTNS
PARAMETER   ANS1=0.0,ANS2=0.0
*       SYSTEM INPUT
        Y1=STEP(T1 )
        Y2=STEP(T2 )
        Y3=STEP(T3)
```

TABLE XXV - Continued

```
        Y4=STEP(T4)
        TT=-5.*(Y1*(TIME-T1)-Y2*(TIME-T2)-Y3*(TIME-T3)+Y4*(TIME-T4))*PLS
*       PILOT  ASSIST SYSTEM
        BIS2=CFB*(CQ*Q1+CTHET*THET1) +CUFB*(CUDT*UDT1+CU*U2)+TT*CTST2
        BIS=REALPL(0.0,TP ,BIS2)
        BIS1=INTGRL(0.0,CPWO*BIS)
        B1C1P=CQ1*Q1
        B1C2P=REALPL(0.0,TS,B1C1P)
        B1C1=B1C1P-B1C2P
        B1C=BIS+BIS1
        B1=B1C+B1C1+TT*CTST1
        P1=CTST5*TT+H
        P5=TLH*P1
        P2=DERIV(0.0,P5)
        P3=P1+P2
        P4=DELAY(1,DP,P3)
        COB2=CFBH*(CH*H+CHDT*HDT+CH2DT*CAZ)+TT*CTST4+CPH*P4
        COB=REALPL(0.0,TC,COB2)
        COB1=INTGRL(0.0,CCWO*COB)
        CO=COB+COB1+TT*CTST3
        PS1=B1C*B1C
        PS2=INTGRL(0.0,PS1)
        PS3=SQRT(PS2)
        SS1=B1C1*B1C1
        SS2=INTGRL(0.0,SS1)
        SS3=SQRT(SS2)
PARAMETER CTST5=0.0
PARAMETER TP=.10,TC=.1
PARAMETER CQ1=0.0
PARAMETER TS=.10
PARAMETER CPWO=0.5,CCWO=0.10
PARAMETER  CFBH=0.0,CH2DT=.16,CHDT=.08,CH=.00
PARAMETER  CFB =0.0,CQ=+.256,CTHET=+.256
PARAMETER  CTST1=1.0,CTST2=0.0,CTST3=0.0,CTST4=0.0
PARAMETER TLH=0.0,DP=0.0,CPH=0.0
PARAMETER CUFB=0.0,CUDT=.10,CU=.017
PARAMETER  PLS=0.5,T1=.02,T2=.22,T3=1.22,T4=1.42
TITLE      FREE A/C RESPONSE TO 0.5 INCH AFT CYCLIC PULSE
TIMER DELT=.020,FINTIM=0.6,PRDEL=.20,OUTDEL=.20
PRINT Q1,THET1,CAZ ,HDT,H,B1,U1,CO
METHOD  RECT
END
TITLE      FREE A/C RESPONSE TO 5.0FT/SEC RMS U GUST
PARAMETER CTST1=0.0,ANS2=1.0
END
TITLE      ATTITUDE HOLD RESPONSE TO 5 DEG PITCH ATT CMD
PARAMETER ANS2=0.0,CFB=1.0,CTST2=1.0,PLS=1.25
END
TITLE      ATTITUDE HOLD RESPONSE TO 5.0 FT/SEC RMS U GUST
PARAMETER CTST2=0.0,ANS2=1.0,PLS=0.5
```

TABLE XXV - Continued

```
          END
          TITLE      AIRSPEED RESPONSE TO 5.0 FT/SEC AIRSPEED CMD
          PARAMETER ANS2=0.0,CTST2=1.0
          END
          TITLE      VERTICAL RATE RESPONSE TO 5.0 FT/SEC UP VERT RATE CMD
          PARAMETER CFBH=1.0,CTST2=0.0,PLS=60.0
          END
          TITLE      ATTITUDE HOLD RESP TO 5 FT/SEC RMS U GUST WITH SER/PAR SERVO
          END
          STOP

OUTPUT VARIABLE SEQUENCE
ALFAG  UG     Y4     Y3     Y2     Y1     TT     CO     Q      Q1
B1C1P  B1C1   B1C    B1     UDT    U      ALFADT ALFA   THE2DT ZZ0007
THEDT  THETA  U1     HDT    H      X      NOISE  ZZ0016 FLINS  U2
UDT1   THET1  BIS2   ZZ0019 BIS    ZZ0022 BIS1   ZZ0025 B1C2P  P1
P5     P2     P3     P4     CA2    COB2   ZZ0028 COB    ZZ0031 COB1
PS1    PS2    SS1    SS2    PS3    SS3

 OUTPUTS    INPUTS    PARAMS   INTEGS + MEM BLKS   FORTRAN   DATA CDS
 60(500)  133(1400)  58(400)   14+   1= 15(300)   58(600)      44
```

TABLE XXVI. SIMPLIFIED LATERAL CSMP PROGRAM LISTING

```
        ****CONTINUOUS SYSTEM MODELING PROGRAM****

     ***PROBLEM INPUT STATEMENTS***

TITLE        JOB 72 SIMPLIFIED LAT CSMP   LAT1
*       AIRCRAFT EQUATIONS
        BETADT    =YV*BETA+UYP*P+GUO*PHI+UYR*R+UYA1*A1+UYDLR*DLR...
                  +YV*BETAG
        PHI2DP    =ALV*BETA+ULP*P+UIXZ*RDTP+ULR*R+ULA1*A1+ULDLR*DLR...
                  +ALV*BETAG+ULP*PG
        RDT       =ANV*BETA+UNP*P+UIZZ*PHI2DT+ANR*R+UNA1*A1+UNDLR*DLR...
                  +ANV*BETAG+UNP*PG
        PHI2DT=KIX*PHI2DP
        RDTP=REALPL(0.0,TRDT,RDT)
        BETA      =INTGRL(0.0,BETADT)
        PHIDT     =INTGRL(0.0,PHI2DT)
        PHI       =INTGRL(0.0,PHIDT)
        P         =PHIDT
        R         =INTGRL(0.0,RDT)
        PSI=INTGRL(0.0,R)
        R1        =57.3*R
        PHI1      =57.3*PHI
        AY=U0*(YV*BETA+UYDLR*DLR)+ALY*RDTP
        AY1=U0*(YV*BETA+UYDLR*DLR)
        UY=INTGRL(0.0,AY1)
        Y=INTGRL(0.0,UY)
        P1        =57.3*P
        PSI1=57.3*PSI
PARAMETER    YV=-.52    ,UYP=-.1270 ,GUO=+.237   ,UYR=-.986
PARAMETER    UYA1=+.0063,UYDLR=+.0135
PARAMETER    ALV=-14.2   ,ULP=-3.98 ,UIXZ=+1.41  ,ULR=+2.86
PARAMETER    ULA1=+1.83 ,ULDLR=+3.12
PARAMETER    ANV=+5.42   ,UNP=-.082 ,UIZZ=+.132  ,ANR=-1.39
PARAMETER    UNA1=-.0012,UNDLR=-1.37
PARAMETER    KIX=1.00,TRDT=.04
PARAMETER U0=135.8,ALY=0.0
*
*              GUST INPUT
*
        NOISE=GAUSS(1,0.0,STD)
        FLTNS=REALPL(0.0,TNS,NOISE)
PARAMETER    STD=5.0,TNS=3.18
        BETAG=BNS1*FLTNS
        PG=BNS2*FLTNS
PARAMETER    BNS1=0.0,BNS2=0.0
*        PILOT ASSIST SYSTEM
        R2PP=YW0*R1
```

TABLE XXVI - Continued

```
      R2P=REALPL(0.0,TYW0,R2PP)
       R1P=R1-R2P
      DRSC3=CAYY*AY
      DRSC4=REALPL(0.0,TAY,DRSC3)
      DRSC1=CYOB*(CPSI*PSI1+DRSC4-CR*R1P)
      DRSC2=CTST8*TT+DRSC1
      DRSC=REALPL(0.0,TY,DRSC2)
      D1=INTGRL(0.0,CYW0*DRSC)
      DLP=DRSC+D1+CTST7*TT
PARAMETER  YW0=0.0,TYW0=1.0
      VIL=CFIB*(CP*P1+CPHI*PHI1)
      VOL=CROB*(CAY*AY+CUY*UY)
      VSUM=-VIL-VOL-CTST6*TT
      VSUMP=REALPL(0.0,TR,VSUM)
      A2=INTGRL(0.0,CRW0*VSUMP)
      A1=A2+VSUMP*CTST5*TT
PARAMETER  CROB=0.0,CFIB=0.0,CYOB=0.0
PARAMETER  CUY=0.0,CAY=0.0,CPHI=.032,CP=.128
PARAMETER  CTST5=0,CTST7=0,CTST6=0,CTST8=0
PARAMETER  CPSI=0.0,CR=2.5,CAYY=0.0
PARAMETER   TAY=2.0
PARAMETER   CRW0=0.1,CYW0=0.1
PARAMETER   TR=.1,TY=.1
*     SYSTEM INPUT
      Y1=STEP(T1 )
      Y2=STEP(T2 )
      Y3=STEP(T3)
      Y4=STEP(T4)
      TT=-5.*(Y1*(TIME-T1)-Y2*(TIME-T2)-Y3*(TIME-T3)+Y4*(TIME-T4))*PLS
PARAMETER  PLS=0.5,T1=.02,T2=.22,T3=1.22,T4=1.42
TITLE     FREE A/C RESPONSE TO 0.5 INCH RIGHT CYCLIC PULSE
TIMER DELT=.020,FINTIM=.2,PRDEL=.10,OUTDEL=.10
PRINT  P1,R1,PHI1,A1,DLP,AY,UY
METHOD  RECT
END
TITLE     FREE A/C RESPONSE TO 5.0 FT/SEC RMS SIDE GUST
END
TITLE     ATTITUDE HOLD RESPONSE TO 5.0 DEG ROLL ATT CMD
END
TITLE     ATTITUDE HOLD RESPONSE TO 5.0 FT/SEC RMS SIDE GUST
END
TITLE     AIRSPEED RESP TO 5.0 FT/SEC AIRSPEED CMD
END
STOP

OUTPUT VARIABLE SEQUENCE
PG     BETAG  Y4     Y3     Y2     Y1     TT     DLR    A1     P
PHI2DP PHI2DT RDT    ZZ0003 RDTP   BETADT BETA   PHID1  PHI    R
PSI    AY1    UY     Y      NOISE  ZZ0020 FLTNS  R1     R2PP   ZZ0023
R2P    AY     DRSC3  ZZ0026 DRSC4  R1P    PSI1   DRSC1  DRSC2  ZZ0029
```

TABLE XXVI - Continued

```
DRSC    220032 D1       V01     PH11    P1        VIL     VSUM    270035 VSUMR
220035 A2

 OUTPUTS     INPUTS     PARAMS  INTEGS + MEM BLKS  FORTRAN  DATA CDS
 56(500)   129(1400)   56(400)   15+   0= 15(300)    55(600)      33
```

TABLE XXVII. COMPLEX LONGITUDINAL CSMP PROGRAM LISTING

```
TITLE           JOB 72 LONG 3DOF COMPEX PAS LONG1
*      AIRCRAFT EQUATIONS %LONG<
       UDT      #XU*U&XW*ALFA&UXQ*Q-GUO*THETA&UXB1*B1&UXCO*CO
       ALFADT   #ZU*U&ZW*ALFA&UZQ*Q-GUOS*THETA&UZB1*B1&UZCO*CO
       THE2DT   #AMU*U&AMW*ALFA&UMQ*Q&UMB1*B1&UMCO*CO
        U       #INTGRL%0.0,UDT<
       ALFA     #INTGRL%0.0,ALFADT<
       THEDT    #INTGRL%0.0,KIY*THE2DT<
       THETA    #INTGRL%0.0,THEDT<
       Q        #THEDT
       CAZ      #U1*%Q-ALFADT<&ALX*THE2DT
       HDT      #U1*%THETA-ALFA<&HDTIC
       H        #INTGRL%0.0,HDT<
       UDT1#UO*UDT
       U1#UO*%1.0&U<
       U2#UO*U
       Q1       #57.3*Q
       THET1    #57.3*THETA
PARAMETER   KIY#.800
PARAMETER   XU#-.046,XW#-.061,UXQ#&.122,GUO#&.237
PARAMETER   UXB1#&.012,UXCO#-.01
PARAMETER   ZU#-.087,ZW#-1.13,UZQ#&.980,GUOS#-.027
PARAMETER   UZB1#&.04,UZCO#-.161
PARAMETER   AMU#-.377,AMW#1.965,UMQ#-.513
PARAMETER   UMB1#-.313,UMCO#&.320
PARAMETER   HDTIC#0.0,UO#135.8,ALX#0.0
*      SYSTEM INPUT
       Y1#STEP%T1 <
       Y2#STEP%T2 <
       Y3#STEP%T3<
       Y4#STEP%T4<
       TT#-5.*%Y1*%TIME-T1<-Y2*%TIME-T2<-Y3*%TIME-T3<&Y4*%TIME-T4<<*PLS
*       SLO IN/OUT
       VEL#NOT%ATT<
       ATT#PULSE%TWT1,T10<
PARAMETER T10#0.0,TWT1#25.0
       FS1#FCNSW%ATT,0.0,0.0,1.0<
       FS2#FCNSW%VEL,0.0,0.0,1.0<
       FS11#REALPL%0.0,TS1,FS1<
       FS12#REALPL%0.0,TS2,FS2<
       FS21#FCNSW%NSLO1,FS11,FS11,1.0<
       FS22#FCNSW%NSLO1,FS12,FS12,1.0<
PARAMETER   TS1#0.5,TS2#0.5
       NORM#NOT%ALTHLD<
       ALTHLD#PULSE%TWT2,T11<
PARAMETER T11#0.0,TWT2#25.0
       FS3#FCNSW%ALTHLD,0.0,0.0,1.0<
       FS4#FCNSW%NORM,0.0,0.0,1.0<
       FS13#REALPL%0.0,TS3,FS3<
```

TABLE XXVII - Continued

```
       FS14#REALPL%0.0,TS4,FS4<
       FS23#FCNSW%NSL02,FS13,FS13,1.0<
       FS24#FCNSW%NSL02,FS14,FS14,1.0<
PARAMETER  TS3#0.5,TS4#0.5
PARAMETER  NSL01#1.0,NSL02#1.0
*       PITCH PAS
       THP1#REALPL%0.0,TWO,THET1<
       THET10#THET1-THP1
       THET6#-THET1&THS1
       THET7#FCNSW%SW1,0.0,0.0,THET6<
       THET8#FCNSW%VEL,0.0,0.0,THET6<
       THET9#THET7&THET8
       THS1#INTGRL%0.0,-40.0*THET9<
       THS2#REALPL%0.0,TH,THS1<
       THET3#THET1-THS2
       THET4#CTHET*THET3&CTHETW*THET10
       THET5#THET4*FS21
       THA#CPROT*%THET5&CQ*Q1<
       UDTP1#REALPL%0.0,UFIL,UDT1<
       UP1#-U2&UP3
       UP7#FCNSW%ATT,0.0,0.0,UP1<
       UP8#FCNSW%SW1,0.0,0.0,UP1<
       UP2#UP7&UP8
       UP3#INTGRL%0.0,-40.0*UP2<
       UP4#REALPL%0.0,TU,UP3<
       UP9#U2-UP4
       UP5#CU*UP9&CUDT*UDTP1
       UP6#UP5*FS22
       FCYC#CTST3*TT
       FYC1#DEADSP%F1,F2,FCYC<
       FYC2#CFP2*FYC1
       FYC3#REALPL%0.0,TCP2,FYC2<
       FYC4#FYC3-UP6*CLONG
       FYC5#LIMIT%F3,F4,FYC4<
       B1R#CTST2*TT  &FYC5&THA
       FYC16#CFP3*FYC1&.02*FYC7&FYC9
       FYC7#INTGRL%0.0,-10.0*FYC16<
       FYC8#ABS%FYC7<
       FYC9#FCNSW%FTR,FYC10,FYC10,0.0<
       FYC10# 20.0*FYC7
       FYC11#DEADSP%F5,F6,FCYC<
       FYC12#ABS%FYC11<
       FTR #COMPAR%FYC12,FT2<
       SW1#COMPAR%FYC8,FT1<
PARAMETER  FT1#1.9,FT2#.15
PARAMETER  CFP2#.6,CFP3#1.0,TCP2#.2
PARAMETER  F1#-1.5,F2#1.5,F5#-1.5,F6#1.5,F3#-15.0,F4#15.0
*       PITCH SERVO
       B1RP#INTGRL%0.0,CPW0*B1R<
       B1C#B1R&B1RP
```

```
      B1#B1C&CTST1*TT
PARAMETER  CPWO#0.5
*     COLL PAS
      H2DTP1#REALPL%0.0,.04,CAZ  <
      HP1#-H &HP4
      HP3#FCNSW%ALTHLD,HP1,HP1,0.0<
      HP4#INTGRL%0.0,-40.0*HP3<
      HP5#REALPL%0.0,TL2,HP4<
      HDT1#CH*%H-HP5<&CHDTD*HDT
      HDT2#HDT1*FS23
      FZ8# CALT*%HDT2&CH2DT*H2DTP1<
      HDTP1#-HDT&HDTP4
      HDTP2#FCNSW%SW11,0.0,0.0,HDTP1<
      HDTP7#CRS1*HDTP4
      HDTP3#FCNSW%SW12,HDTP7,HDTP7,HDTP2<
      HDTP4#INTGRL%0.0,-40.0*HDTP3<
      HDTP5#REALPL%0.0,TL1,HDTP4<
      HDTP6#ABS%HDT<
      SW12#COMPAR%HDTP6,FT4<
      FCOLL#CTST5*TT
      FZ1#DEADSP%F9,F10,FCOLL<
      FZ2#CFC1*FZ1
      FZ3#REALPL%0.0,TCP1,FZ2<
      FZ20#CHDT*%HDT-HDTP5<
      FZ21#FZ20*FS24
      FZ30#FCNSW%SW11,FCOLL,FCOLL,0.0<
      FZ31#CFC2*FZ30
      FZ32#REALPL%0.0,TCP2,FZ31<
      FZ6#FZ3&FZ32 -FZ21*CVERT
      FZ7#LIMIT%F7,F8,FZ6<
      FZ4#DEADSP%F11,F12,FCOLL<
      FZ5#ABS%FZ4<
      SW11#COMPAR%FZ5,FT3<
      COR#CTST4*TT  &FZ7-FZ8
PARAMETER  F7#-15.0,F8#15.0,F9#-1.5,F10#1.5,F11#-1.5,F12#1.5
PARAMETER  FT3#1.5,FT4#1.5
PARAMETER  CFC1#.6,CFC2#.06,TCP1#.2,TCP2#.2
*       COLL SERVO
      CORP#INTGRL%0.0,CCWC*COR<
      COC#COR&CORP
      CO#COC&CTST6*TT
PARAMETER  CCWO#0.1
TITLE  FREE A/C RESPONSE TO &0.5 INCH PITCH INPUT
PARAMETER  PLS#5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PRINT Q1,THET1,CAZ ,HDT,H,B1,U1,CO
METHOD  RECT
PARAMETER  CTST1#1.0,CTST2#0.0,CTST3#0.0
PARAMETER  CTST6#0.0,CTST4#0.0,CTST5#0.0
PARAMETER  CPROT#0.0,CLONG#0.0
```

TABLE XXVII - Continued

```
PARAMETER                 CQ#.256,CTHET#.256,TH#.5,CTHETW#0.0,TWO#2.0
PARAMETER  CU#.017,CUDT#.10,TU#.5,UFIL#.04
PARAMETER  CVERT#0.0,CALT#0.0
PARAMETER                 CH2DT#.16,CHDTD#.08,CH#.008,TL2#.5
PARAMETER  CHDT#0.0,CRS1#1.0,TL1#.5
END
TITLE  FREE A/C RESPONSE TO &0.5 INCH COLL  INPUT
PARAMETER  PLS#5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PARAMETER  CTST1#0.0,CTST2#0.0,CTST3#0.0
PARAMETER  CTST6#1.0,CTST4#0.0,CTST5#0.0
END
TITLE   PITCH RATE RESPONSE TO 0.5 INCH SERVO CMD W/O IN
PARAMETER  PLS#5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PARAMETER  CPROT#1.0,CLONG#0.0
PARAMETER                 CQ#.256,CTHET#.000,TH#.5,CTHETW#0.0,TWO#2.0
END
TITLE PITCH VEL RESPONSE TO 2.0 LB FORCE INPUT W/O IN
PARAMETER  PLS#20.0,T1#.00,T2#.02,T3#2.15,T4#2.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PARAMETER  CPROT#1.0,CLONG#1.0
PARAMETER  CU#.017,CUDT#.10,TU#.5,UFIL#.04
PARAMETER  CTST1#0.0,CTST2#0.0,CTST3#1.0
END
TITLE PITCH ATT RESPONSE TO 2.0 LB FORCE INPUT W/O IN
PARAMETER  PLS#20.0,T1#.00,T2#.02,T3#2.15,T4#2.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PARAMETER  CPROT#1.0,CLONG#0.0
PARAMETER                 CQ#.256,CTHET#.256,TH#.5,CTHETW#0.0,TWO#2.0
END
TITLE   VERT ACCEL RESPONSE TO 0.5 INCH SERVO CMD W/O IN
PARAMETER  PLS# 5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PARAMETER                 CQ#.256,CTHET#.000,TH#.5,CTHETW#0.0,TWO#2.0
PARAMETER  CPROT#1.0,CLONG#1.0
PARAMETER  CVERT#0.0,CALT#1.0
PARAMETER                 CH2DT#.16,CHDTD#.00,CH#.000,TL2#.5
PARAMETER  CTST1#0.0,CTST2#0.0,CTST3#0.0
PARAMETER  CTST6#0.0,CTST4#1.0,CTST5#0.0
END
TITLE NORM MODE RESPONSE %LOW RATE,LOW FORCE< TO 1.0 LB FORCE
PARAMETER  PLS# 5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
PARAMETER  CVERT#1.0,CALT#1.0
PARAMETER  CHDT#.08,CRS1#1.0,TL1#.5
PARAMETER  CTST6#0.0,CTST4#0.0,CTST5#1.0
PARAMETER  PLS#10.0,T1#.00,T2#.02,T3#2.15,T4#2.17
END
TITLE NORM MODE RESPONSE %LOW RATE,HI  FORCE< TO 2.0 LB FORCE
```

TABLE XXVII - Continued

```
PARAMETER  PLS#20.0,T1#.00,T2#.02,T3#2.15,T4#2.17
TIMER DELT#.001,FINTIM#3.0,PRDEL#.20,OUTDEL#.20
END
TITLE NORM MODE RESPONSE %HI  RATE,HI  FORCE< TO 2.0 LB FORCE
PARAMETER  PLS#20.0,T1#.00,T2#.02,T3#2.15,T4#2.17
PARAMETER  HDTIC#5.0,U0#135.8
END
TITLE NORM MODE RESPONSE %HI  RATE,LOW FORCE< TO 1.0 LB FORCE
PARAMETER  PLS#10.0,T1#.00,T2#.02,T3#2.15,T4#2.17
END
TITLE ALTITUDE HOLD RESPONSE TO 2.0 LB FORCE INPUT
PARAMETER  PLS#20.0,T1#.00,T2#.02,T3#2.15,T4#2.17
PARAMETER  HDTIC#0.0,U0#135.8
PARAMETER  CVERT#0.0,CALT#1.0
PARAMETER            CH2DT#.16,CHDTD#.08,CH#.003,TL2#.5
END
TITLE  PITCH MODE SW RESP WITH 0.5 IN PITCH INPUT
PARAMETER  PLS#5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
PARAMETER  NSLO1#0.0,NSLO2#1.0
PARAMETER T10#1.0,TWT1#25.0
PARAMETER T11#0.0,TWT2#25.0
PARAMETER  CTST1#0.0,CTST2#1.0,CTST3#0.0
PARAMETER  CTST6#0.0,CTST4#0.0,CTST5#0.0
PARAMETER W0#1.0,CFU#1.0,CFUW0#1.0,TPW0#2.0,CFUL#.0582
PARAMETER  CCW0#0.1
PARAMETER  CPROT#1.0,CLONG#1.0
PARAMETER            CQ#.256,CTHET#.256,TH#.5,CTHETW#0.0,TW0#2.0
PARAMETER  CU#.017,CUDT#.10,TU#.5,UFIL#.04
PARAMETER  CVERT#0.0,CALT#0.0
PARAMETER            CH2DT#.16,CHDTD#.08,CH#.008,TL2#.5
PARAMETER  CHDT#0.0,CRS1#1.0,TL1#.5
END
TITLE  VERT MODE SW RESP WITH 0.5 IN COLL INPUT
PARAMETER  PLS#5.0,T1#.00,T2#.02,T3#1.15,T4#1.17
PARAMETER  NSLO1#1.0,NSLO2#0.0
PARAMETER T10#0.0,TWT1#25.0
PARAMETER T11#1.0,TWT2#25.0
PARAMETER  CTST1#0.0,CTST2#0.0,CTST3#0.0
PARAMETER  CTST6#0.0,CTST4#1.0,CTST5#0.0
END
STOP
ENDJOB
```

TABLE XXVIII. SIMPLIFIED 6-DOF CSMP PROGRAM LISTING

```
****CONTINUOUS SYSTEM MODELING PROGRAM****

***PROBLEM INPUT STATEMENTS***

*
TITLE  SIX DEGREE OF FREEDOM UH-1B SIMULATION
*
*      AIRCRAFT EQUATIONS
       UDT      =XU*U+XW*ALFA+UXQ*Q-GUO*THETA+UXB1*B1+UXCO*CO ...
       +XV*BETA+UXP*P+UXR*R+UXA1*A1+UXDLR*DLR
       ALFADT   =ZU*U+ZW*ALFA+UZQ*Q-GUOS*THETA+UZB1*B1+UZCO*CO ...
       +ZV*BETA+UZP*P+UZR*R+UZA1*A1+UZDLR*DLR
       THE2DT   =AMU*U+AMW*ALFA+UMQ*Q+UMB1*B1+UMCO*CO   ...
       +AMV*BETA+UMP*P+UMR*R+UMA1*A1+UMDLR*DLR
       BETADT   =YV*BETA+UYP*P+GUO*PHI+UYR*R+UYA1*A1+UYDLR*DLR ...
       +YU*U+YW*ALFA+UYQ*Q+UYB1*B1+UYCO*CO
       PHI2DP   =ALV*BETA+ULP*P+UIXZ*RDTP+ULR*R+ULA1*A1+ULDLR*DLR ...
       +ALU*U+ALW*ALFA+ULQ*Q+ULB1*B1+ULCO*CO
       RDT      =ANV*BETA+UNP*P+UIZZ*PHI2DT+ANR*R+UNA1*A1+UNDLR*DLR...
       +ANU*U+ANW*ALFA+ANQ*Q+UNB1*B1+UNCO*CO
       PHI2DT=KIX*PHI2DP
        U       =INTGRL(0.0,UDT)
       ALFA     =INTGRL(0.0,ALFADT)
       THEDT    =INTGRL(0.0,KIY*THE2DT)
       THETA    =INTGRL(0.0,THEDT)
       Q        =THEDT
       BETA     =INTGRL(0.0,BETADT)
       PHIDT    =INTGRL(0.0,PHI2DT)
       R        =INTGRL(0.0,RDT)
       P        =PHIDT
       RDTP=REALPL(0.0,TRDT,RDT)
         PHI=INTGRL(0.0,PHIDT)
PARAMETER    KIX=1.00,KIY=.800
PARAMETER    XU=-.048,XW=-.081,UXQ=+.122,GUO=+.237
PARAMETER    UXB1=+.012,UXCO=-.01
PARAMETER   XV=.0124,UXP=-.004,UXR=-.0004,UXA1=-.00004,UXDLR=-.00035
PARAMETER    ZU=-.087,ZW=-1.13,UZQ=+.980,GUOS=-.027
PARAMETER    UZB1=+.04,UZCO=-.161
PARAMETER   ZV=-.056,UZP=-.017,UZR=+.0015,UZA1=-.0006,UZDLR=.00018
PARAMETER    AMU=-.377,AMW=1.965,UMQ=-.513
PARAMETER    UMB1=-.313,UMCO=+.320
PARAMETER   AMV=.082,UMP=.0902,UMR=.0197,UMA1=-.0002,UMDLR=.0306
PARAMETER    YV=-.52    ,UYP=-.1270 ,GUO=+.237   ,UYR=-.986
PARAMETER    UYA1=+.0063,UYDLR=+.0135
PARAMETER   YU=.0007,YW=-.0394,UYQ=-.0021 ,UYB1=.00176,UYCO=-.0057
PARAMETER    ALV=-14.2   ,ULP=-3.98 ,UIXZ=+1.41  ,ULR=+2.86
PARAMETER    ULA1=+1.83 ,ULDLR=+3.12
```

TABLE XXVIII - Continued

```
PARAMETER  ALU=-.025,ALW=-9.4,ULQ=-.542 ,ULB1=.438,ULCO=-1.36
PARAMETER   ANV=+5.42  ,UNP=-.082 ,UIZZ=+.132 ,ANR=-1.39
PARAMETER   UNA1=-.0012,UNDLR=-1.37
PARAMETER  ANU=-.77,ANW=-1.13,ANQ=+.271  ,UNB1=.038,UNCO=.537
PARAMETER  U0=135.8,W=6750.,WO=-15.2,TRDT=.04
      B1RP=UMP1*P
      B1R=TT*CTST2+B1RP
      B1C1=CPITCH*B1R
      B1C=REALPL(0.0,TP,B1C1)
      B1=TT*CTST1+B1C
      CORP=0.0
      COR=TT*CTST4+CORP
      COC=REALPL(0.0,TC,COR)
      CO=TT*CTST3+COC
      COP=ULCO1*CO
      COP1=REALPL(0.0,TXCR,COP)
      COP2=COP-COP1
      DLRP=ULDLR1*DLR
      DLRP1=REALPL(0.0,TXDR,DLRP)
      DLRP2=DLRP-DLRP1
      A1RP=ULQ1*Q+ULR1*R+COP2+DLRP2
      A1R=TT*CTST6+A1RP
      A1C1=CROLL*A1R
      A1C=REALPL(0.0,TR,A1C1)
      A1=TT*CTST5+A1C
      COP3=UNCO1*CO
      COP4=REALPL(0.0,TXCY,COP3)
      COP5=COP3-COP4
      DLRRP=COP5
       DLRR=TT*CTST8+DLRRP
      DLRC1=CYAW*DLRR
      DLRC=REALPL(0.0,TY,DLRC1)
      DLR=TT*CTST7+DLRC
PARAMETER   CPITCH=0.0,UMP1=+.288
PARAMETER   CROLL=0.0,ULQ1=+.296,ULR1=-.56,ULCO1=+.446,ULDLR1=-1.26
PARAMETER   CYAW=0.0,UNCO1=+.39
PARAMETER  TXCR=1.0,TXDR=1.0,TXCY=1.0
PARAMETER  CTST2=0.0,TP=.03,CTST1=1.0
PARAMETER  CTST4=0.0,TC=.03,CTST3=0.0
PARAMETER  CTST6=0.0,TR=.03,CTST5=0.0
PARAMETER  CTST8=0.0,TY=.03,CTST7=0.0
*
*     SYSTEM INPUTS
*
      Y1=STEP(T1 )
      Y2=STEP(T2 )
      Y3=STEP(T3)
      Y4=STEP(T4)
      TT=-5.*(Y1*(TIME-T1)-Y2*(TIME-T2)-Y3*(TIME-T3)+Y4*(TIME-T4))*PLS
PARAMETER  PLS=20.0,T1=.00,T2=.02,T3=2.15,T4=2.17
```

TABLE XXVIII - Continued

```
          TITLE     FREE A/C RESPONSE TO .5 INCH AFT CYCLIC
          METHOD  RECT
          PRINT  P,Q,R,A1,DLR,B1,CO
          TIMER DELT=.001,FINTIM=0.2,PRDEL=.20,OUTDEL=.20
          END
          PARAMETER   CPITCH=1.0,UMP1=+.288
          END
          TITLE     FREE A/C RESPONSE TO .5 INCH RIGHT CYCLIC
          PARAMETER  CPITCH=0.0,UMP1=+.288
          END
          STOP

OUTPUT VARIABLE SEQUENCE
Y4      Y3      Y2      Y1      TT      DLP     A1      P       CO      B1
Q       UDT     U       ALFADT  ALFA    THE2DT  ZZ0007  THEDT   THETA   BETADT
BETA    PHI2DP  PHI2DT  PHIDT   RDT     R       ZZ0018  RDTP    PHI     B1RP
B1R     B1C1    ZZ0023  B1C     CORP    COR     ZZ0026  COC     COP     ZZ0029
COP1    DLRP    ZZ0032  DLRP1   DLRP2   COP2    A1RP    A1P     A1C1    ZZ0035
A1C     COP3    ZZ0038  COP4    COP5    DLRRP   DLRR    DLRC1   ZZ0041  DLRC

 OUTPUTS     INPUTS    PARAMS  INTEGS + MEM BLKS  FORTRAN  DATA CDS
 54(500)   184(1400) 103(400)   16+   0= 16(300)   67(600)      40
```

TABLE XXIX. MINOR LOOP CSMP PROGRAM LISTING

```
*
TITLE  TORQUE MOTOR SERVO LOOP
*
      B1R#CTST1*TT&CTST2*TSIN
      F11#B1R-%CTG*F20&CTAC*F16&F23-F25<
      F12#CSA*F11
      F50#LIMIT%ALMT1,ALMT2,F12<
      F13#F50-F14-CBEMF*F18
      F14#CB01*F15
      F15#SIGN%1.0,F18<
      F16#CM*F13
      F30#LIMIT%TLMT1,TLMT2,F16<
      F17#F30-CSP*F22-CB02*F31-CTS*F18-CTST3*TT-CTST4*TSIN
      F18#INTGRL%0.0,CIN*F17<
      F19P#CCL*F13
      F19#REALPL%0.0,TCL,F19P<
      F20#DEADSP%CLMT1,CLMT2,F19<
      F21#INTGRL%0.0,.573*F18<
      F22#HSTRSS%0.0,BKLSH1,BKLSH2,F21<
      F31#SIGN%1.0,F22<
      B1C#CFUL*F22
      F23#CFU*B1C
      F24P#CFUVO*B1C
      F24#REALPL%0.0,TPVO,F24P<
      F25#FCNSW%VO,0.0,0.0,F24<
PARAMETER   ALMT1#-4.0,ALMT2#4.0
PARAMETER CTG#.77,CTAC#.00095,CSA#33.6,CM#.584,CCL#1.38,CBEMF#.0069
PARAMETER  CSP#.0000,TCL#.005
PARAMETER CLMT1#-3.75,CLMT2#3.75,BKLSH1#-.05,BKLSH2#.05
PARAMETER CB01#0.14,CB02#0.0,TLMT1#-2.5,TLMT2#2.5,CIN#2910.0
PARAMETER VO#0.0,CFU#1.0,CFUVO#1.0,TPVO#2.0,CFUL#.292
*
*     SYSTEM INPUTS
*
      Y1#STEP%T1 <
      Y2#STEP%T2 <
      Y3#STEP%T3<
      Y4#STEP%T4<
      TT#-5.*%Y1*%TIME-T1<-Y2*%TIME-T2<-Y3*%TIME-T3<&Y4*%TIME-T4<<*PLS
PARAMETER  PLS#50.00,T1#.00,T2#.02,T3#2.15,T4#2.17
      TSIN#SINE%DLY,OMEGA,SHIFT<
PARAMETER OMEGA#0.10,DLY#0.0,SHIFT#0.0
PARAMETER CTST1#1.0,CTST2#0.0,CTST3#0.0,CTST4#0.0
PARAMETER CTS#0.0
TITLE  MINOR LOOP STEP RESPONSE TO 5 DEG STEP CMD NO V/O
METHOD  RECT
PRINT BIR,BIC,F18,F16,F30,F20,F11
TIMER DELT#.001,FINTIM#0.6,PRDEL#.10,OUTDEL#.10
```

TABLE XXIX - Continued

```
END
STOP
 ENDJOB
EOF:
>FILE

11.03.04 >
```

TABLE XXX. MINOR LOOP CSMP PROGRAM EXECUTION

```
*** CSMP/360 SIMULATION DATA ***

TITLE  TORQUE MOTOR SERVO LOOP

PARAMETER   ALMT1=-4.0,ALMT2=+4.0

PARAMETER CTG=22.0,CTAC=.00246,CSA=16.9,CM=.584,CCL=1.05,CBEMF=.0182

PARAMETER  CSP=.063,TCL=.005

PARAMETER CLMT1=-3.75,CLMT2=3.75,BKLSH1=-.05,BKLSH2=.05

PARAMETER CBO1=0.14,CBO2=0.0,TLMT1=-2.5,TLMT2=2.5,CIN=8330.0

PARAMETER WO=0.0,CFU=1.0,CFUWO=1.0,TPWO=2.0,CFUL=.128

PARAMETER  PLS=12.10,T1=.00,T2=.02,T3=2.15,T4=2.17

PARAMETER OMEGA=0.10,DLY=0.0,SHIFT=0.0

PARAMETER CTST1=.316,CTST2=0.0,CTST3=0.0,CTST4=0.0

PARAMETER CTS=.0081

TITLE   PITCH MINOR LOOP STEP RESPONSE TO 3 DEG STEP CMD NO W/O

METHOD  RECT

 PRINT B1R,F22,F18,F50, F11

TIMER DELT=.001,FINTIM=.3,PRDEL=.025,OUTDEL=.025

END


TIMER VARIABLES
  DELT   = 1.0000E-03
 DELMIN= 3.0000E-08
 FINTIM= 3.0000E-01
 PRDEL  = 2.5000E-02
 OUTDEL= 2.5000E-02
```

TABLE XXX - Continued

TORQUE MOTOR SERVO LOOP
RECT INTEGRATION
PITCH MINOR LOOP STEP RESPONSE TO 3 DEG STEP CMD NO W/O

TIME	B1R	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
2.5000E-02	-3.8236E-01	-4.8559E-01	-7.3046E 01	-2.3746E 00	-1.4051E-01
5.0000E-02	-3.8236E-01	-1.3460E 00	-4.6768E 01	-1.6058E 00	-9.5020E-02
7.5000E-02	-3.8236E-01	-1.8855E 00	-2.9218E 01	-1.1685E 00	-6.9140E-02
1.0000E-01	-3.8236E-01	-2.2225E 00	-1.8255E 01	-8.9524E-01	-5.2973E-02
1.2500E-01	-3.8236E-01	-2.4331E 00	-1.1405E 01	-7.2452E-01	-4.2871E-02
1.5000E-01	-3.8236E-01	-2.5646E 00	-7.1255E 00	-6.1788E-01	-3.6561E-02
1.7500E-01	-3.8236E-01	-2.6468E 00	-4.4520E 00	-5.5125E-01	-3.2618E-02
2.0000E-01	-3.8236E-01	-2.6981E 00	-2.7817E 00	-5.0062E-01	-3.0155E-02
2.2500E-01	-3.8236E-01	-2.7302E 00	-1.7381E 00	-4.8361E-01	-2.8616E-02
2.5000E-01	-3.8236E-01	-2.7503E 00	-1.0863E 00	-4.6737E-01	-2.7655E-02
2.7500E-01	-3.8236E-01	-2.7628E 00	-6.7891E-01	-4.5721E-01	-2.7054E-02
3.0000E-01	-3.8236E-01	-2.7706E 00	-4.2449E-01	-4.5088E-01	-2.6679E-02

*** CSMP/360 SIMULATION DATA ***

TITLE MINOR LOOP STEP RESP TO 6 DEG CMD NO WO

PARAMETER PLS=24.2

END

TIMER VARIABLES
DELT = 1.0000E-03
DELMIN= 3.0000E-08
FINTIM= 3.0000E-01
PRDEL = 2.5000E-02
OUTDEL= 2.5000E-02

MINOR LOOP STEP RESP TO 6 DEG CMD NO WO
RECT INTEGRATION

TIME	B1R	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
2.5000E-02	-7.6472E-01	-8.9380E-01	-1.1382E 02	-4.0000E 00	-3.7032E-01
5.0000E-02	-7.6472E-01	-2.5165E 00	-1.0114E 02	-3.2753E 00	-1.9381E-01
7.5000E-02	-7.6472E-01	-3.6843E 00	-6.3260E 01	-2.3238E 00	-1.3750E-01

TABLE XXX - Continued

1.0000E-01	-7.6472E-01	-4.4141E 00	-3.9522E 01	-1.7322E 00	-1.0250E-01
1.2500E-01	-7.6472E-01	-4.8699E 00	-2.4692E 01	-1.3626E 00	-8.0528E-02
1.5000E-01	-7.6472E-01	-5.1548E 00	-1.5426E 01	-1.1317E 00	-6.6964E-02
1.7500E-01	-7.6472E-01	-5.3327E 00	-9.6379E 00	-9.8743E-01	-5.8428E-02
2.0000E-01	-7.6472E-01	-5.4439E 00	-6.0217E 00	-8.9731E-01	-5.3095E-02
2.2500E-01	-7.6472E-01	-5.5133E 00	-3.7623E 00	-8.4100E-01	-4.9763E-02
2.5000E-01	-7.6472E-01	-5.5567E 00	-2.3508E 00	-8.0582E-01	-4.7682E-02
2.7500E-01	-7.6472E-01	-5.5838E 00	-1.4689E 00	-7.8384E-01	-4.6381E-02
3.0000E-01	-7.6472E-01	-5.6007E 00	-9.1795E-01	-7.7011E-01	-4.5569E-02

*** CSMP/360 SIMULATION DATA ***

TITLE MINOR LOOP STEP RESP TO 12 DEG CMD NO WO

PARAMETER PLS=48.4

END

TIMER VARIABLES
DELT = 1.0000E-03
DELMIN= 3.0000E-08
FINTIM= 3.0000E-01
PRDEL = 2.5000E-02
OUTDEL= 2.5000E-02

MINOR LOOP STEP RESP TO 12 DEG CMD NO WO
RECT INTEGRATION

TIME	B1R	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
2.5000E-02	-1.5294E 00	-1.0742E 00	-1.1543E 02	-4.0000E 00	-1.1080E 00
5.0000E-02	-1.5294E 00	-2.7189E 00	-1.1258E 02	-4.0000E 00	-9.0447E-01
7.5000E-02	-1.5294E 00	-4.2949E 00	-1.0726E 02	-4.0000E 00	-7.1585E-01
1.0000E-01	-1.5294E 00	-5.7958E 00	-1.0214E 02	-4.0000E 00	-5.3630E-01
1.2500E-01	-1.5294E 00	-7.2253E 00	-9.7275E 01	-4.0000E 00	-3.6531E-01
1.5000E-01	-1.5294E 00	-8.5833E 00	-8.8506E 01	-3.6007E 00	-2.1306E-01
1.7500E-01	-1.5294E 00	-9.6095E 00	-5.5630E 01	-2.7475E 00	-1.6258E-01
2.0000E-01	-1.5294E 00	-1.0251E 01	-3.4755E 01	-2.2273E 00	-1.3179E-01
2.2500E-01	-1.5294E 00	-1.0652E 01	-2.1713E 01	-1.9023E 00	-1.1256E-01
2.5000E-01	-1.5294E 00	-1.0903E 01	-1.3566E 01	-1.6992E 00	-1.0054E-01
2.7500E-01	-1.5294E 00	-1.1059E 01	-8.4756E 00	-1.5724E 00	-9.3039E-02
3.0000E-01	-1.5294E 00	-1.1157E 01	-5.2953E 00	-1.4931E 00	-8.8348E-02

TABLE XXX - Continued

*** CSMP/360 SIMULATION DATA ***

TITLE MINOR LOOP RESP AT .1 CPS

PARAMETER OMEGA=.628

TIMER DELT=.001,FINTIM=20.0,PRDEL=.5,OUTDEL=.5

END

TIMER VARIABLES
DELT = 1.0000E-03
DELMIN= 2.0000E-06
FINTIM= 2.0000E 01
PRDEL = 5.0000E-01
OUTDEL= 5.0000E-01

MINOR LOOP RESP AT .1 CPS
RECT INTEGRATION

TIME	B1R	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
5.0000E-01	1.1830E-01	7.2136E-01	3.0014E 00	3.1396E-01	1.8578E-02
1.0000E 00	2.2502E-01	1.5290E 00	2.5901E 00	3.8778E-01	2.2945E-02
1.5000E 00	3.0975E-01	2.1810E 00	1.9248E 00	4.3669E-01	2.5840E-02
2.0000E 00	3.6418E-01	2.6138E 00	1.0714E 00	4.5595E-01	2.6979E-02
2.5000E 00	3.8300E-01	2.7849E 00	1.1326E-01	4.4366E-01	2.6252E-02
3.0000E 00	3.6437E-01	2.8186E 00	-1.3574E 00	1.1705E-01	6.9259E-03
3.5000E 00	3.1010E-01	2.4259E 00	-1.7418E 00	6.5507E-02	3.8761E-03
4.0000E 00	2.2552E-01	1.8196E 00	-2.4563E 00	-2.2729E-02	-1.3449E-03
4.5000E 00	1.1888E-01	1.0414E 00	-2.9310E 00	-1.2180E-01	-7.2072E-03
5.0000E 00	6.1132E-04	1.6736E-01	-3.1197E 00	-2.2201E-01	-1.3137E-02
5.5000E 00	-1.1771E-01	-7.1699E-01	-3.0028E 00	-3.1353E-01	-1.8552E-02
6.0000E 00	-2.2453E-01	-1.5252E 00	-2.5928E 00	-3.8746E-01	-2.2927E-02
6.5000E 00	-3.0939E-01	-2.1782E 00	-1.9287E 00	-4.3652E-01	-2.5829E-02
7.0000E 00	-3.6399E-01	-2.6122E 00	-1.0760E 00	-4.5593E-01	-2.6978E-02
7.5000E 00	-3.8300E-01	-2.7847E 00	-1.1821E-01	-4.4380E-01	-2.6260E-02
8.0000E 00	-3.6456E-01	-2.8187E 00	1.3182E 00	-1.1834E-01	-7.0021E-03
8.5000E 00	-3.1046E-01	-2.4284E 00	1.7377E 00	-6.5910E-02	-3.9000E-03
9.0000E 00	-2.2601E-01	-1.8231E 00	2.4532E 00	2.2247E-02	1.3164E-03
9.5000E 00	-1.1946E-01	-1.0456E 00	2.9293E 00	1.2128E-01	7.1765E-03
1.0000E 01	-1.2226E-03	-1.7191E-01	3.1195E 00	2.2151E-01	1.3107E-02
1.0500E 01	1.1713E-01	7.1262E-01	3.0041E 00	3.1311E-01	1.8527E-02
1.1000E 01	2.2403E-01	1.5214E 00	2.5956E 00	3.8714E-01	2.2908E-02
1.1500E 01	3.0902E-01	2.1754E 00	1.9326E 00	4.3634E-01	2.5819E-02
1.2000E 01	3.6380E-01	2.6107E 00	1.0807E 00	4.5591E-01	2.6977E-02
1.2500E 01	3.8300E-01	2.7846E 00	1.2317E-01	4.4394E-01	2.6269E-02

TABLE XXX - Continued

2.4000E-01	3.6380E-01	1.2046E-01	7.8858E 01	2.6092E 00	1.5439E-01
2.5000E-01	3.8300E-01	5.5788E-01	7.1873E 01	2.2778E 00	1.3478E-01
2.6000E-01	3.6474E-01	9.3651E-01	5.7802E 01	1.7352E 00	1.0268E-01
2.7000E-01	3.1082E-01	1.2190E 00	3.8031E 01	1.0348E 00	6.1233E-02
2.8000E-01	2.2650E-01	1.3775E 00	1.4500E 01	2.4529E-01	1.4514E-02
2.9000E-01	1.2004E-01	1.4060E 00	-7.3956E 00	-7.0528E-01	-4.1733E-02
3.0000E-01	1.8343E-03	1.3925E 00	-3.3382E 01	-1.5934E 00	-9.4285E-02
3.1000E-01	-1.1655E-01	1.1418E 00	-5.5020E 01	-2.1522E 00	-1.2735E-01
3.2000E-01	-2.2354E-01	7.8411E-01	-7.0313E 01	-2.5507E 00	-1.5093E-01
3.3000E-01	-3.0866E-01	3.5602E-01	-7.8724E 01	-2.7137E 00	-1.6057E-01
3.4000E-01	-3.6361E-01	-1.0064E-01	-7.9443E 01	-2.6245E 00	-1.5529E-01
3.5000E-01	-3.8299E-01	-5.4126E-01	-7.2400E 01	-2.2918E 00	-1.3561E-01
3.6000E-01	-3.6493E-01	-9.2277E-01	-5.8281E 01	-1.7482E 00	-1.0344E-01
3.7000E-01	-3.1118E-01	-1.2079E 00	-3.8467E 01	-1.0468E 00	-6.1939E-02
3.8000E-01	-2.2700E-01	-1.3688E 00	-1.4894E 01	-2.5608E-01	-1.5153E-02
3.9000E-01	-1.2062E-01	-1.3986E 00	7.0359E 00	6.9444E-01	4.1091E-02
4.0000E-01	-2.4449E-03	-1.3880E 00	3.3014E 01	1.5887E 00	9.4005E-02
4.1000E-01	1.1597E-01	-1.1389E 00	5.4782E 01	2.1461E 00	1.2699E-01
4.2000E-01	2.2304E-01	-7.8249E-01	7.0133E 01	2.5463E 00	1.5067E-01
4.3000E-01	3.0830E-01	-3.5529E-01	7.8602E 01	2.7110E 00	1.6042E-01
4.4000E-01	3.6341E-01	1.0083E-01	7.9378E 01	2.6235E 00	1.5524E-01
4.5000E-01	3.8299E-01	5.4121E-01	7.2387E 01	2.2924E 00	1.3564E-01
4.6000E-01	3.6511E-01	9.2277E-01	5.8312E 01	1.7500E 00	1.0355E-01
4.7000E-01	3.1153E-01	1.2082E 00	3.8532E 01	1.0495E 00	6.2098E-02
4.8000E-01	2.2749E-01	1.3695E 00	1.4981E 01	2.5925E-01	1.5340E-02
4.9000E-01	1.2120E-01	1.3996E 00	-6.9381E 00	-6.9090E-01	-4.0882E-02
5.0000E-01	3.0562E-03	1.3898E 00	-3.2906E 01	-1.5868E 00	-9.3892E-02
5.1000E-01	-1.1538E-01	1.1413E 00	-5.4713E 01	-2.1441E 00	-1.2687E-01
5.2000E-01	-2.2254E-01	7.8513E-01	-7.0091E 01	-2.5453E 00	-1.5061E-01
5.3000E-01	-3.0793E-01	3.5808E-01	-7.8595E 01	-2.7112E 00	-1.6043E-01
5.4000E-01	-3.6322E-01	-9.8080E-02	-7.9408E 01	-2.6250E 00	-1.5532E-01

MINOR LOOP SINE RESP AT 5 CPS
RECT INTEGRATION

TIME	B1R	F22	F18	F50	F11
5.5000E-01	-3.8298E-01	-5.3873E-01	-7.2452E 01	-2.2949E 00	-1.3580E-01
5.6000E-01	-3.6530E-01	-9.2075E-01	-5.8409E 01	-1.7535E 00	-1.0376E-01
5.7000E-01	-3.1189E-01	-1.2068E 00	-3.8652E 01	-1.0535E 00	-6.2337E-02
5.8000E-01	-2.2798E-01	-1.3688E 00	-1.5114E 01	-2.6353E-01	-1.5593E-02
5.9000E-01	-1.2178E-01	-1.3994E 00	7.1631E 00	6.7138E-01	3.9726E-02
6.0000E-01	-3.6730E-03	-1.3886E 00	3.2766E 01	1.5794E 00	9.3456E-02

TABLE XXX - Continued

*** CSMP/360 SIMULATION DATA ***

```
TITLE MINOR LOOP SINE RESP AT 5 CPS

PARAMETER CTST1=0.0,CTST2=.383

PARAMETER OMEGA=31.4

TIMER DELT=.001,FINTIM=.60,PRDEL=.01,OUTDEL=.01

END
```

TIMER VARIABLES
DELT = 1.0000E-03
DELMIN= 6.0000E-08
FINTIM= 6.0000E-01
PRDEL = 1.0000E-02
OUTDEL= 1.0000E-02

MINOR LOOP SINE RESP AT 5 CPS
RECT INTEGRATION

TIME	B1R	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
1.0000E-02	1.1830E-01	0.0	1.7938E 01	1.2534E 00	7.4167E-02
2.0000E-02	2.2502E-01	1.4840E-01	4.0337E 01	1.8049E 00	1.0680E-01
3.0000E-02	3.0975E-01	4.1832E-01	5.3826E 01	2.0920E 00	1.2379E-01
4.0000E-02	3.6418E-01	7.4344E-01	5.8709E 01	2.1057E 00	1.2460E-01
5.0000E-02	3.8300E-01	1.0744E 00	5.5098E 01	1.8579E 00	1.0994E-01
6.0000E-02	3.6437E-01	1.3644E 00	4.3819E 01	1.3847E 00	8.1935E-02
7.0000E-02	3.1010E-01	1.5729E 00	2.6368E 01	7.4205E-01	4.3908E-02
8.0000E-02	2.2552E-01	1.6697E 00	4.7725E 00	8.7637E-04	5.1856E-05
9.0000E-02	1.1888E-01	1.6740E 00	-1.5488E 01	-9.6822E-01	-5.7291E-02
1.0000E-01	6.1168E-04	1.5909E 00	-4.0858E 01	-1.7326E 00	-1.0252E-01
1.1000E-01	-1.1771E-01	1.3036E 00	-6.0583E 01	-2.2906E 00	-1.3554E-01
1.2000E-01	-2.2453E-01	9.1671E-01	-7.4881E 01	-2.6644E 00	-1.5766E-01
1.3000E-01	-3.0939E-01	4.6467E-01	-8.2454E 01	-2.8058E 00	-1.6603E-01
1.4000E-01	-3.6399E-01	-1.1474E-02	-8.2462E 01	-2.6983E 00	-1.5966E-01
1.5000E-01	-3.8300E-01	-4.6779E-01	-7.4818E 01	-2.3503E 00	-1.3907E-01
1.6000E-01	-3.6456E-01	-8.6183E-01	-6.0199E 01	-1.7939E 00	-1.0615E-01
1.7000E-01	-3.1046E-01	-1.1569E 00	-3.9977E 01	-1.0823E 00	-6.4041E-02
1.8000E-01	-2.2601E-01	-1.3255E 00	-1.6081E 01	-2.8369E-01	-1.6786E-02
1.9000E-01	-1.1946E-01	-1.3597E 00	6.4402E 00	6.5470E-01	3.8740E-02
2.0000E-01	-1.2230E-03	-1.3541E 00	3.2098E 01	1.5740E 00	9.3135E-02
2.1000E-01	1.1713E-01	-1.1094E 00	5.4108E 01	2.1299E 00	1.2603E-01
2.2000E-01	2.2403E-01	-7.5658E-01	6.9545E 01	2.5315E 00	1.4979E-01
2.3000E-01	3.0902E-01	-3.3262E-01	7.8061E 01	2.6967E 00	1.5957E-01

TABLE XXX - Continued

*** CSMP/360 SIMULATION DATA ***

TITLE MINOR LOOP SINE RESP AT 1 CPS

PARAMETER OMEGA=6.28

TIMER DELT=.001,FINTIM=2.0,PRDEL=.10,OUTDEL=.10

END

TIMER VARIABLES
DELT = 1.0000E-03
DELMIN= 2.0000E-07
FINTIM= 2.0000E 00
PRDEL = 1.0000E-01
OUTDEL= 1.0000E-01

MINOR LOOP SINE RESP AT 1 CPS
RECT INTEGRATION

TIME	[illegible]	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
1.0000E-01	2.2502E-01	8.3496E-01	2.3903E 01	1.0030E 00	5.9346E-02
2.0000E-01	3.6418E-01	2.0973E 00	1.7371E 01	8.9557E-01	5.2993E-02
3.0000E-01	3.6437E-01	2.6434E 00	7.0855E-01	4.1005E-01	2.4264E-02
4.0000E-01	2.2552E-01	2.3110E 00	-1.7211E 01	-4.7233E-01	-2.7949E-02
5.0000E-01	6.1097E-04	9.7895E-01	-2.7979E 01	-9.4413E-01	-5.5866E-02
6.0000E-01	-2.2453E-01	-6.9190E-01	-2.8436E 01	-1.1156E 00	-6.6014E-02
7.0000E-01	-3.6399E-01	-2.0735E 00	-1.8097E 01	-9.1368E-01	-5.4064E-02
8.0000E-01	-3.6456E-01	-2.6401E 00	-8.6071E-01	-4.1417E-01	-2.4507E-02
9.0000E-01	-2.2601E-01	-2.3138E 00	1.7186E 01	4.7121E-01	2.7882E-02
1.0000E 00	-1.2215E-03	-9.8307E-01	2.7965E 01	9.4329E-01	5.5816E-02
1.1000E 00	2.2403E-01	6.8775E-01	2.8449E 01	1.1157E 00	6.6015E-02
1.2000E 00	3.6380E-01	2.0708E 00	1.8134E 01	9.1464E-01	5.4121E-02
1.3000E 00	3.6474E-01	2.6399E 00	9.0805E-01	4.1567E-01	2.4596E-02
1.4000E 00	2.2650E-01	2.3154E 00	-1.7121E 01	-4.6907E-01	-2.7756E-02
1.5000E 00	1.8325E-03	9.8700E-01	-2.7945E 01	-9.4231E-01	-5.5758E-02
1.6000E 00	-2.2354E-01	-6.8362E-01	-2.8462E 01	-1.1157E 00	-6.6017E-02
1.7000E 00	-3.6361E-01	-2.0682E 00	-1.8172E 01	-9.1558E-01	-5.4177E-02
1.8000E 00	-3.6493E-01	-2.6398E 00	-9.5528E-01	-4.1718E-01	-2.4685E-02
1.9000E 00	-2.2700E-01	-2.3182E 00	1.7092E 01	4.6787E-01	2.7684E-02
2.0000E 00	-2.4449E-03	-9.91[illegible]0E-01	2.7931E 01	9.4143E-01	5.5706E-02

TABLE XXX - Continued

1.3000E 01	3.6474E-01	2.8186E 00	-1.2724E 00	1.1981E-01	7.0894E-03
1.3500E 01	3.1082E-01	2.4309E 00	-1.7335E 00	6.6314E-02	3.9239E-03
1.4000E 01	2.2650E-01	1.8267E 00	-2.4502E 00	-2.1769E-02	-1.2881E-03
1.4500E 01	1.2004E-01	1.0499E 00	-2.9276E 00	-1.2077E-01	-7.1461E-03
1.5000E 01	1.8339E-03	1.7645E-01	-3.1193E 00	-2.2102E-01	-1.3078E-02
1.5500E 01	-1.1655E-01	-7.0824E-01	-3.0055E 00	-3.1268E-01	-1.8502E-02
1.6000E 01	-2.2354E-01	-1.5176E 00	-2.5984E 00	-3.8683E-01	-2.2889E-02
1.6500E 01	-3.0866E-01	-2.1726E 00	-1.9364E 00	-4.3616E-01	-2.5808E-02
1.7000E 01	-3.6361E-01	-2.6091E 00	-1.0854E 00	-4.5589E-01	-2.6976E-02
1.7500E 01	-3.8299E-01	-2.7844E 00	-1.2817E-01	-4.4408E-01	-2.6277E-02
1.8000E 01	-3.6493E-01	-2.8185E 00	1.2270E 00	-1.2127E-01	-7.1759E-03
1.8500E 01	-3.1118E-01	-2.4335E 00	1.7294E 00	-6.6716E-02	-3.9477E-03
1.9000E 01	-2.2700E-01	-1.8308E 00	2.4469E 00	2.1290E-02	1.2597E-03
1.9500E 01	-1.2062E-01	-1.0542E 00	2.9258E 00	1.2026E-01	7.1160E-03
2.0000E 01	-2.4460E-03	-1.8100E-01	3.1191E 00	2.2053E-01	1.3049E-02

*** CSMP/360 SIMULATION DATA ***

TITLE MINOR LOOP STEP RESP TO 3 DEG STEP CMD WITH W/O

PARAMETER CTST1=.316,CTST2=0.0,W0=1.0

TIMER DELT=.001,FINTIM=3.0,PRDEL=.1,OUTDEL=.1

END

TIMER VARIABLES
DELT = 1.0000E-03
DELMIN= 3.0000E-07
FINTIM= 3.0000E 00
PRDEL = 1.0000E-01
OUTDEL= 1.0000E-01

MINOR LOOP STEP RESP TO 3 DEG STEP CMD WITH W/O
RECT INTEGRATION

TIME	B1R	F22	F18	F50	F11
0.0	0.0	0.0	0.0	0.0	-0.0
1.0000E-01	-1.5294E 00	-5.7958E 00	-1.0214E 02	-4.0000E 00	-5.5320E-01
2.0000E-01	-1.5294E 00	-1.0493E 01	-4.3087E 01	-2.5201E 00	-1.4912E-01
3.0000E-01	-1.5294E 00	-1.1885E 01	-1.4043E 01	-1.8531E 00	-1.0965E-01
4.0000E-01	-1.5294E 00	-1.2534E 01	-9.8223E 00	-1.8042E 00	-1.0676E-01
5.0000E-01	-1.5294E 00	-1.3073E 01	-9.1933E 00	-1.8447E 00	-1.0915E-01
6.0000E-01	-1.5294E 00	-1.3596E 01	-9.0844E 00	-1.8979E 00	-1.1230E-01
7.0000E-01	-1.5294E 00	-1.4116E 01	-9.0508E 00	-1.9530E 00	-1.1556E-01

TABLE XXX - Continued

8.0000E-01	-1.5294E 00	-1.4634E 01	-9.0282E 00	-2.0081E 00	-1.1882E-01
9.0000E-01	-1.5294E 00	-1.5150E 01	-9.0070E 00	-2.0632E 00	-1.2208E-01
1.0000E 00	-1.5294E 00	-1.5666E 01	-8.9860E 00	-2.1181E 00	-1.2533E-01
1.1000E 00	-1.5295E 00	-1.6180E 01	-8.9741E 00	-2.1732E 00	-1.2859E-01
1.2000E 00	-1.5295E 00	-1.6693E 01	-8.9639E 00	-2.2282E 00	-1.3184E-01
1.3000E 00	-1.5295E 00	-1.7204E 01	-8.9427E 00	-2.2827E 00	-1.3507E-01
1.4000E 00	-1.5295E 00	-1.7715E 01	-8.9195E 00	-2.3371E 00	-1.3829E-01
1.5000E 00	-1.5295E 00	-1.8225E 01	-8.8959E 00	-2.3913E 00	-1.4149E-01
1.6000E 00	-1.5295E 00	-1.8733E 01	-8.8748E 00	-2.4454E 00	-1.4470E-01
1.7000E 00	-1.5295E 00	-1.9240E 01	-8.8549E 00	-2.4994E 00	-1.4790E-01
1.8000E 00	-1.5295E 00	-1.9746E 01	-8.8342E 00	-2.5534E 00	-1.5109E-01
1.9000E 00	-1.5295E 00	-2.0250E 01	-8.8122E 00	-2.6071E 00	-1.5427E-01
2.0000E 00	-1.5295E 00	-2.0754E 01	-8.7876E 00	-2.6606E 00	-1.5743E-01
2.1000E 00	-1.5295E 00	-2.1256E 01	-8.7706E 00	-2.7142E 00	-1.6060E-01
2.2000E 00	-7.2929E-05	-1.7406E 01	1.8108E 02	4.0000E 00	3.4720E-01
2.3000E 00	-7.2929E-05	-1.1861E 01	3.2449E 01	-2.2511E-01	-1.3320E-02
2.4000E 00	-7.2929E-05	-1.1018E 01	5.0295E 00	-9.0558E-01	-5.3585E-02
2.5000E 00	-7.2929E-05	-1.0877E 01	1.0615E 00	-1.0020E 00	-5.9289E-02
2.6000E 00	-7.2929E-05	-1.0838E 01	4.8570E-01	-1.0139E 00	-5.9996E-02
2.7000E 00	-7.2929E-05	-1.0813E 01	3.9870E-01	-1.0137E 00	-5.9985E-02
2.8000E 00	-7.2929E-05	-1.0791E 01	3.8566E-01	-1.0117E 00	-5.9863E-02
2.9000E 00	-7.2929E-05	-1.0768E 01	3.8382E-01	-1.0094E 00	-5.9726E-02
3.0000E 00	-7.2929E-05	-1.0746E 01	3.8294E-01	-1.0070E 00	-5.9587E-02

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13. ABSTRACT

The purpose of the work performed under this contract was to conduct an analytical investigation of advanced flight control systems (AFCS) requirements for light and medium size helicopters and to design a pilot assist system based on the analytical results. The pilot assist system (PAS) design goal was to develop an AFCS that is relatively light and inexpensive and that can be readily installed in a UH-1B. Some of the significant results of the analytical investigation are as follows.

1. First-cut pilot assist system requirements have been generated.
2. A versatile pilot assist system has been designed for further development and evaluation testing.
3. A math model of the pilot assist system/UH-1B helicopter has been developed.
4. Digital computer simulation and design programs have been developed which can be used to significant advantage in future simulation and flight test work.

Some of the significant results of the pilot assist system design are as follows;

1. A relatively lightweight and inexpensive pilot assist system has been designed;
2. The pilot assist system is flexible (i. e., with respect to control law modification) and should simplify further development and evaluation testing.
3. The system design provides for ease of testing and maintenance.

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Pilot Assist System (PAS)						
Analytical Investigation						
Digital Computer Simulation						

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